CHAPTER 9

FIXED-WING FLIGHT CONTROL SYSTEMS

Chapter Objective: *Upon completion of this chapter, you will have a working knowledge of the functions of fixed-wing flight controls (primary and secondary) and the associated maintenance requirements to include major assembly removal/installation and alignment procedures.*

A flight control system is either a primary or secondary system. Primary flight controls provide longitudinal (pitch), directional (yaw), and lateral (roll) control of the aircraft. Secondary flight controls provide additional lift during takeoff and landing, and decrease aircraft speed during flight, as well as assisting primary flight controls in the movement of the aircraft about its axis. Some manufacturers call secondary flight controls auxiliary flight controls. All systems consist of the flight control surfaces, the respective cockpit controls, connecting linkage, and necessary operating mechanisms.

The systems discussed in this chapter are representative systems. Values such as tolerances, pressures, and temperatures provide better understanding of the text material. You should bear in mind that these values are for representative units and are not accurate for all systems. When actually performing the maintenance procedures discussed, you should consult the current maintenance instruction manual (MIM).

TYPES OF FLIGHT CONTROL SYSTEMS

Learning Objective: *Identify the two basic types of flight control systems.*

A flight control system includes all the components required to control the aircraft about each of the three flight axes. A simple flight control system may be all mechanical; that is, operated entirely through mechanical linkage and cable from the control stick to the control surface. Other more sophisticated flight control systems may use electrical or hydraulic power to provide some or all of the "muscle" in the system. Still others combine all three methods.

MECHANICAL (UNBOOSTED) FLIGHT CONTROL SYSTEM

A typical, simple, mechanical (unboosted) flight control system is the one used in flight training aircraft. The flight control surfaces (ailerons, elevators, and rudder) are moved manually through a series of push-pull rods, cables, bell cranks, sectors, and idlers. Figure 9-1 schematically illustrates the elevator portion of a mechanical (unboosted) flight control system. The control stick is mounted in such a way that it can pivot backwards and forwards on its mounting pin. The control stick is connected to a push-pull rod attached to its lower end. As the stick is moved fore and aft, it causes the elevators to be deflected proportionately.

The push-pull tube (rod) that connects to the lowest point of the control stick extends aft to the pulley. Notice that the function of the pulley is to change the direction of the push-pull action from fore and aft to up and down. The second push-pull tube (rod) connects the forward cable sector and the pulley, and causes the sector to rotate according to the stick movements.

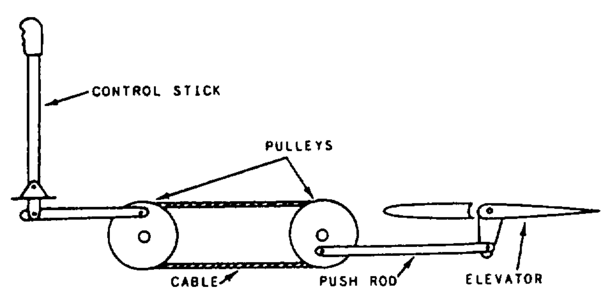


Figure 9-1.—Mechanical (unboosted) flight control system.

From the forward sector, the cables extend back through the aircraft to the aft cable sector. They have been reduced in length so that the remaining essential components of the elevator control system may all be shown in one drawing.

The aft sector is essentially the same as the forward sector, and it acts as a slave to the forward sector. Cables from the forward sector attach to the aft edges of the aft sector. A push-pull tube from the aft sector conncts to the elevator fitting assembly.

The elevator fitting assembly, commonly called the elevator "horn," is built onto the elevators and extends outward (and usually downward) from the elevator surface at right angles to the plane of rotation and the chord line of the elevator surfaces. As the fitting assembly is moved fore or aft, the elevators are moved up or down.

HYDRAULICALLY OPERATED FLIGHT CONTROL SYSTEM

Power-boosted flight control systems are used on high-speed jet aircraft. Aircraft traveling at or near supersonic speeds have such high airloads imposed upon the primary control surfaces that it is impossible for a pilot to control the aircraft without power-operated or power-boosted flight control systems. In the power-boosted system, a hydraulic actuating cylinder is built into the control linkage to assist the pilot in moving the control surface. The power-boost cylinder is still used in the rudder control system of some high-performance aircraft; however, the other primary control surfaces use the full power-operated system. In the full power-operated system, the force necessary to operate the control surface is supplied by hydraulic pressure. Each movable surface is operated by a hydraulic actuator

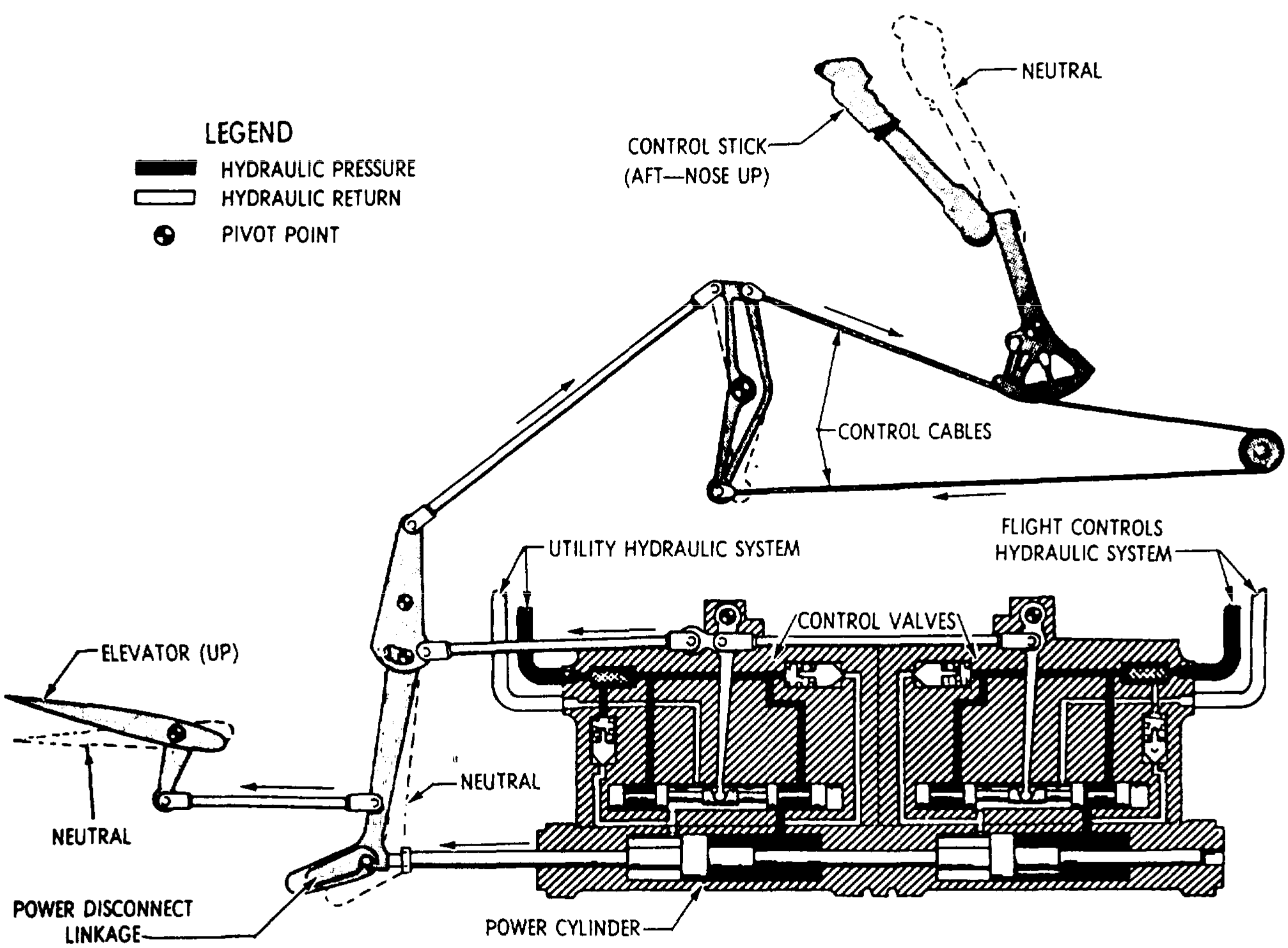


Figure 9-2.—Hydraulically powered elevator control system.

(or power control cylinder) built into the control linkage.

PRIMARY FLIGHT CONTROL SYSTEMS

Learning Objective: *Recognize the functions of the three primary flight control systems (longitudinal, lateral, and directional) and the maintenance associated with each system.*

Different aircraft manufacturers call units of the primary flight control system by a variety of names. The types and complexity of control mechanisms used depend on the size, speed, and mission of the aircraft. A small or low-speed aircraft may have cockpit controls connected directly to the control surface by cables or pushrods. Some aircraft have both cable and a pushrod system. See figure 9-1. The force exerted by the pilot is transferred through them to the control surfaces. On large or high-performance aircraft, the control surfaces have high pressure exerted on them by the airflow. It is difficult for the pilot to move the controls manually. As a result, hydraulic actuators are used within the linkage to aid the pilot in moving the control surface. Figure 9-2 shows a mechanically controlled, hydraulically assisted system. Because these systems reduce pilot fatigue and improve system performance, they are now commonly used. Such systems include automatic pilot, automatic landing systems, and stability augmentation systems.

Navy specifications require two separate hydraulic systems for operating the primary flight control surfaces. Current specifications call for an independent hydraulic power source for emergency operation of the primary flight control surfaces. Some manufacturers provide an emergency system powered by a motor-driven hydraulic pump. Others use a ram-air-driven turbine for operating the emergency system pump.

LONGITUDINAL CONTROL SYSTEMS

Longitudinal control systems control pitch about the lateral axis of the aircraft. Many aircraft use a conventional elevator system for this purpose. Aircraft that operate in the higher speed ranges usually have a movable horizontal stabilizer.

Elevator Control System

The elevator control system, shown in figure 9-2, is typical of many conventional elevator systems. It operates by the control stick in the cockpit and is hydraulically powered.

The operation of the elevator control system starts when the control stick is moved fore or aft. The movement of the stick transfers through the control cables to move the elevator control bell crank. The bell crank transmits the movement to the hydraulic actuating cylinder through the control linkage. The hydraulic actuating cylinder operates a push-pull tube, which deflects the elevators up or down.

The elevator system uses forward and aft bobweights. The bobweights induce a load on the control stick during pitching and vertical acceleration and prevent pilot-induced oscillations through the elevator controls. If the gravity force is increased on the bobweights, the induced load tends to return the control stick to the neutral position. Viscous dampers on the bobweight assemblies retard control stick movement to prevent overcontrol. Overcontrol could cause airframe overstress.

The elevator forward bobweight serves to help recenter the control stick when a heavy gravity load pulls against the airframe. The forward bobweight and damper assembly is in a housing forward of the control stick in the cockpit. See figure 9-3. The

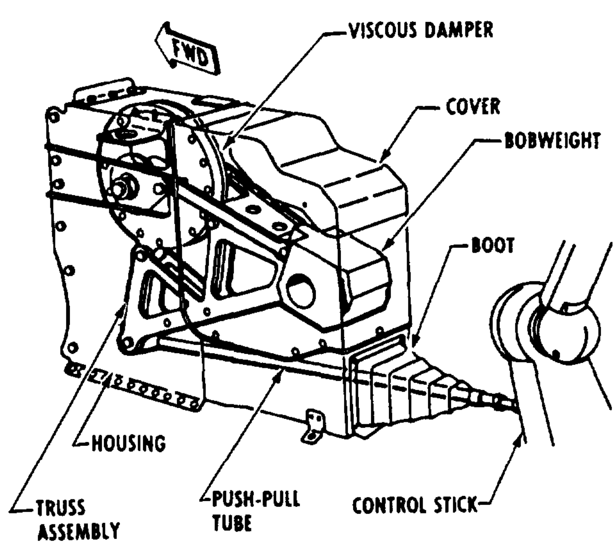


Figure 9-3.—Elevator forward bobweight and damper assembly.

assembly consists of a bobweight, a viscous damper, and a push-pull tube. The push-pull tube is the interconnect between the control stick and the bobweight. The damper is located at the pivot point of the bobweight and restricts fast movement of the bobweight.

The aft bobweight and damper assembly works with the forward assembly to overcome the heavy pull of gravity and retard the chance of overcontrol. See figure 9-4. This assembly is installed in the fuselage, forward and below the horizontal stabilizer. It connects to the elevator control cables.

The aft assembly consists of a bobweight, a viscous damper, and a load spring. The bobweight connects to the elevator control bell crank and the damper. The load spring is between the elevator control bell crank and the fin structure to balance the forward and aft bobweights when the elevator is in a neutral position.

The elevator power mechanism changes the mechanical movement of the control stick to the hydraulic operation of the elevator. See figure 9-5. The mechanism is in the aft section of the aircraft directly below the horizontal stabilizer. As in the aileron power system, the mechanism consists of a hydraulic power cylinder, control valves, linkage, and hydraulic piping.

When the elevator controls are operated, the control valves port hydraulic pressure to the power cylinder. The hydraulic pressure extends or retracts the cylinder piston to move the push-pull tubes. The push-pull tubes deflect the elevators. The control valves are two separate valves connected in tandem by linkage. One valve is supplied hydraulic pressure by the utility hydraulic system. The other valve is supplied hydraulic pressure by the flight control hydraulic system. The power cylinder has dual hydraulic chambers to work from each control valve. Each hydraulic system simultaneously supplies

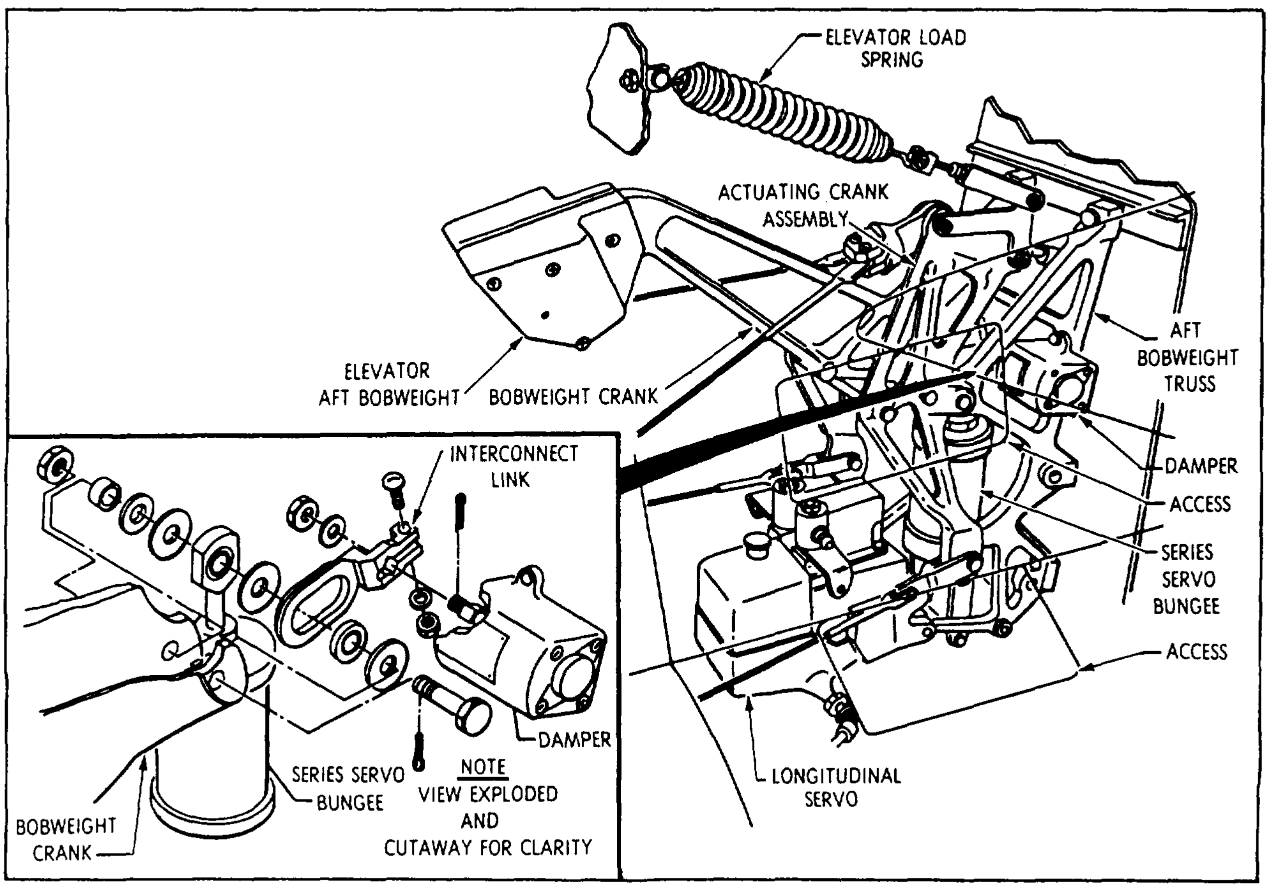


Figure 9-4.—Elevator aft bobweight and damper assembly.

3,000-psi hydraulic pressure to the power mechanism. If one hydraulic systems fails, the other system supplies enough pressure to operate the mechanism. If both hydraulic systems fail, the cylinder disconnects by pulling the MAN FLT CONT (manual flight control) handle in the cockpit. The controls work manually through the linkage of the mechanism to operate the elevators.

The load-feel bungee, shown in figure 9-5, provides an artificial feel to the control stick. The bungee acts as a centering device for the elevator system. Control stick movement compresses the spring in the bungee. Releasing the control stick causes the compressed spring to return the stick to neutral. The bungee also adds a gearing effect between the horizontal stabilizer and the elevators. When the stabilizer is trimmed to give an aircraft nose up condition, the bungee action adds nose up elevator. With the stabilizer trimmed nose down, the bungee action adds nose down attitude on the elevator.

Horizontal Stabilizer Control System (Single Axis)

Various aircraft manufacturers identify the horizontal stabilizer control system by different names. On one aircraft, it is called a unit horizontal tail (UHT) control system. On another aircraft, it is called the stabilizer control system. Regardless of the variation in nomenclature, these systems function to control the aircraft pitch about its lateral axis.

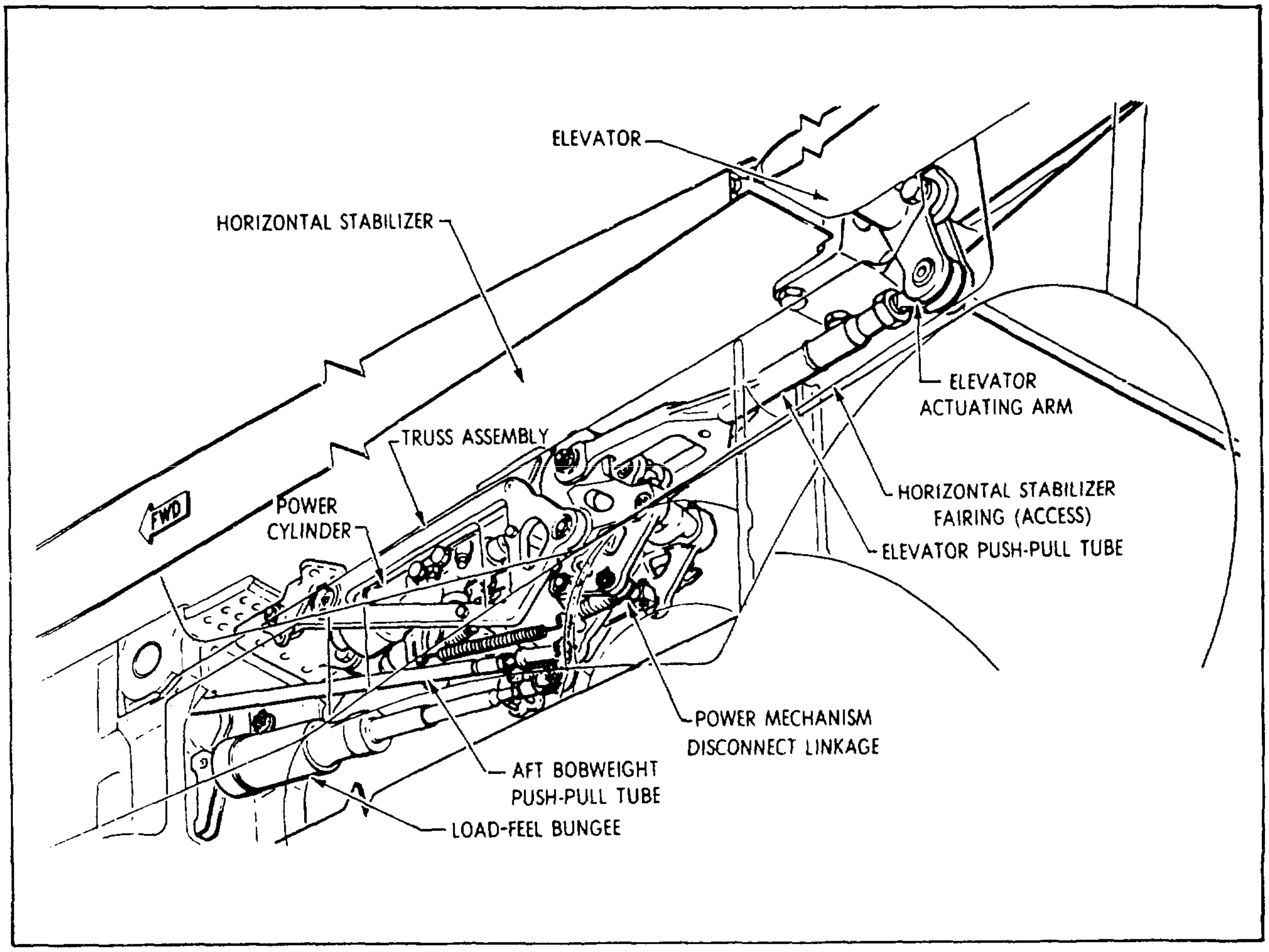
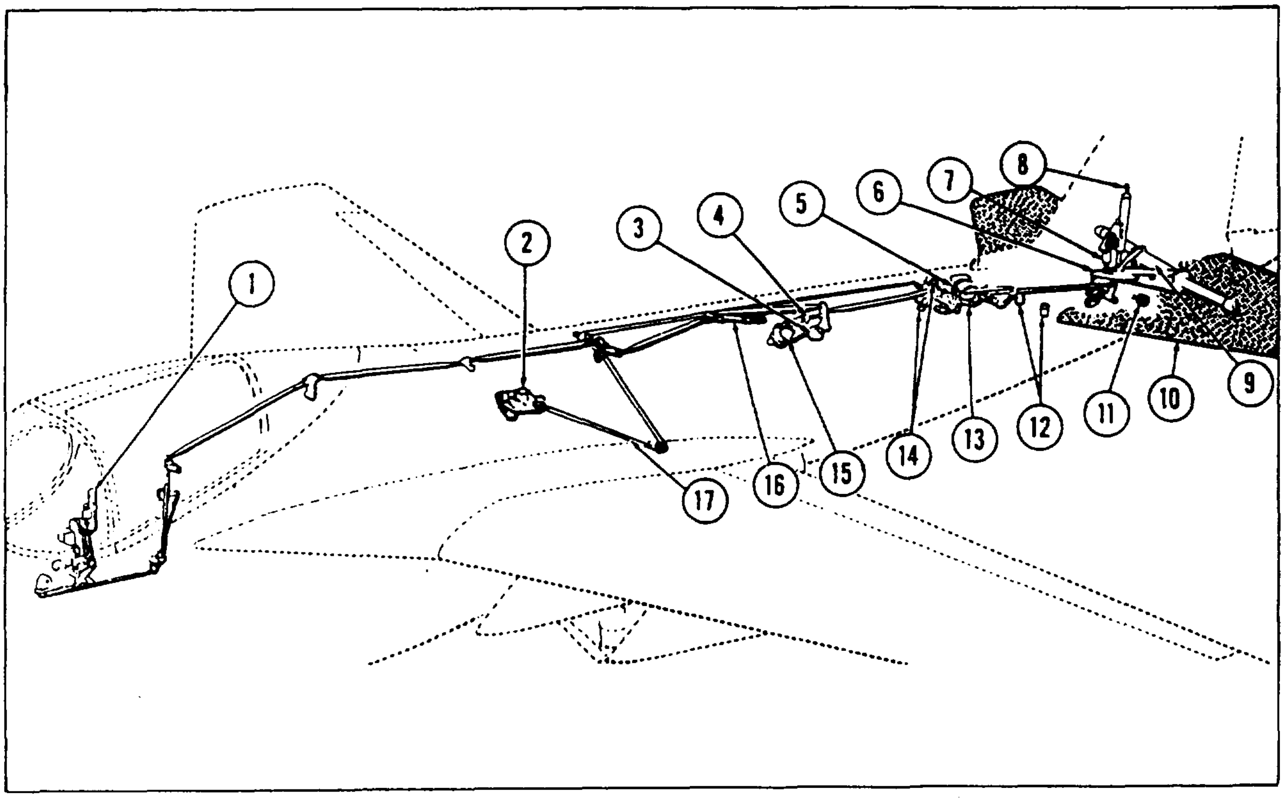


Figure 9-5.—Elevator power mechanism.



1. **Control stick**
2. **Flap drive gearbox**
3. **Trim transmitter**
4. **Artificial feel bungee**
5. **Stabilizer shift mechanism**
6. **Walking beam**
7. **Load-relief bungee**
8. **Stabilizer actuator**
9. **Stabilizer support shaft**
10. **Stabilizer**
11. **Stabilizer position transducer**
12. **Filters**
13. **Negative bobweight**
14. **Clean and dirty switches**
15. **Electrical trim actuator**
16. **Static spring**
17. **Stabilizer shift mechanism cables**

Figure 9-6.–Stabilizer control system.

The horizontal stabilizer control system shown in figure 9-6 is representative of the systems used in many present-day aircraft. The slab-type stabilizer responds to fore-and-aft manual input at the control stick. It responds to automatic flight control system electrical signals introduced at the stabilizer actuator.

In the "clean" configuration, with flaps and slats retracted, stabilizer travel is from 1 1/2 degrees of leading edge up to 10 degrees of leading edge down. In the "dirty" configuration, with flaps and slats extended, stabilizer travel is increased to 24 degrees of leading edge down to provide greater control at slower airspeeds.

Pilot signals are conveyed through bell cranks and pushrods and a trim mechanism to the input linkage of the stabilizer actuator. A trim switch on the control stick grip provides a means of setting stabilizer trim. Stabilizer trim is displayed by the stabilizer trim indicator located on the pilot's lower instrument panel. See figure 9-7.

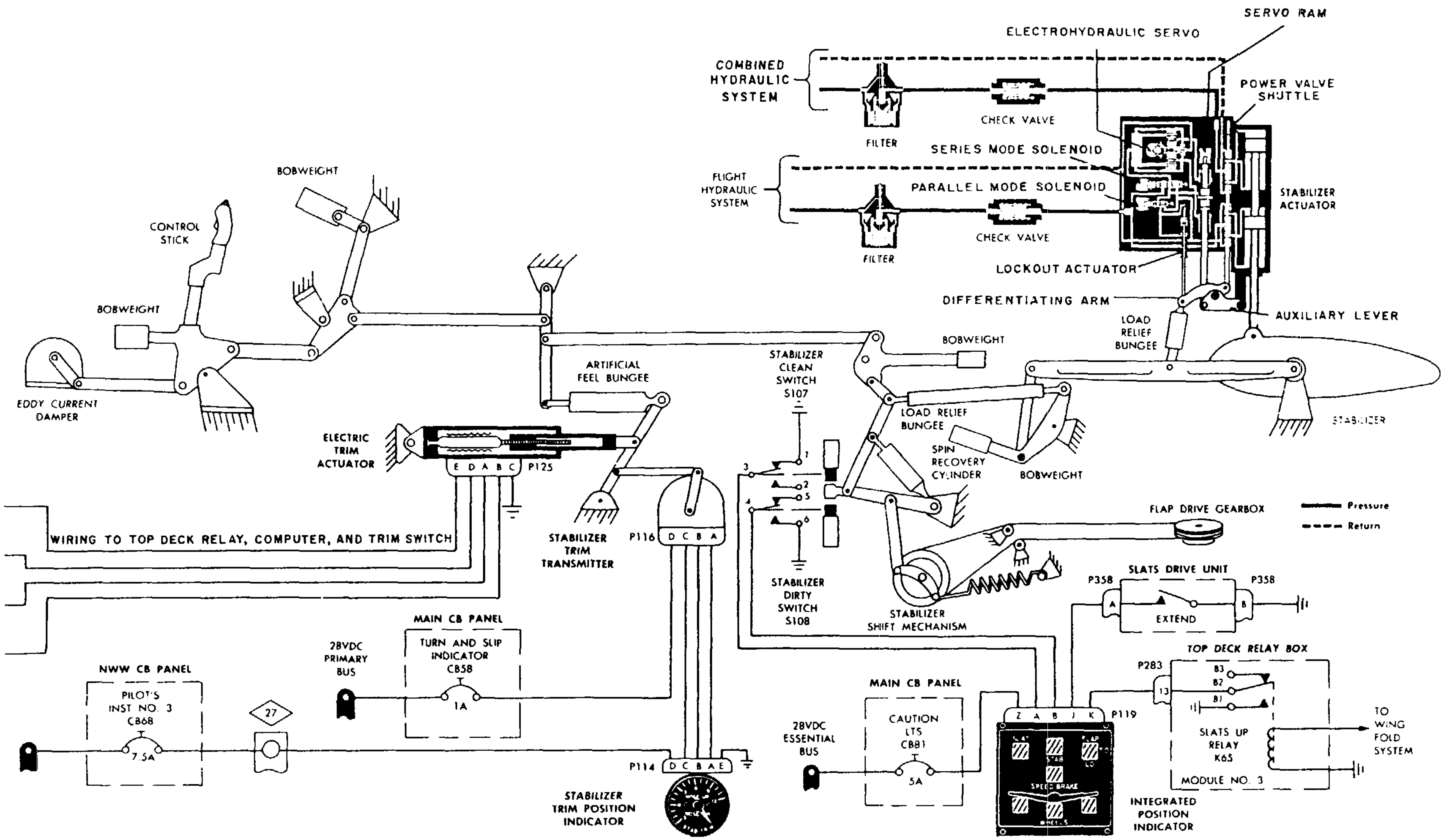


Figure 9-7.—Stabilizer control system schematic.

The position of the stabilizer is shown on the integrated position indicator located on the left side of the pilot's instrument panel. When the stabilizer is in the "clean" configuration, the STAB window of the indicator shows the word CLEAN. When the stabilizer is in the "dirty" configuration, the window shows a picture of a stabilizer.

The stabilizer actuator (fig. 9-7) is a tandem-type actuator powered by both flight and combined system pressures. It contains a power valve shuttle, two tandem-mounted power pistons, a servo ram, an electrohydraulic servo valve, a lockout actuator, and parallel and series mode solenoid valves. The actuator can operate in any of three modes-manual, series, or parallel. Refer to figure 9-7 to help you understand the three modes of operation, as described in the following paragraphs.

MANUAL MODE.—In this mode, the pilot input alone controls the power valve. Inputs are transmitted through linkage to the mechanical input lever. The auxiliary lever is linked in neutral by the servo ram centering springs, causing the mechanical input lever to rotate about its pivot point, moving the power shuttle valve. As the valve shuttle is displaced from neutral, a valve error is established, and pressure is ported to the actuating pistons. The pressure moves the pistons and the attached stabilizer in proportion to the input.

A mechanical feedback is transmitted through the differentiating lever, the load-relief bungee, and the mechanical input lever back to the power valve shuttle, causing it to return to the neutral position.

For a constant velocity pilot input, a small constant valve error is established, and the stabilizer moves at a constant speed. When the pilot input stops, the power shuttle valve is returned to neutral, and the stabilizer stops until a new input is introduced.

SERIES MODE.—In this mode, input signals from the automatic flight control system (AFCS) may be used independently or combined with manual input to control stabilizer movement. The series mode solenoid valve is energized, porting flight system hydraulic pressure to the electrohydraulic servo valve. Input signals from the AFCS amplifier are applied to the coils of a torque motor in the servo valve, regulating flow from the valve to the servo ram.

The servo ram is connected to the auxiliary lever. Movement of the lever moves the mechanical input lever floating-pivot point. This movement causes mechanical input lever rotation about the manual input point and moves the power shuttle valve, causing a valve error.

A linear transduce, mounted on the servo ram center line, provides electrical feedback signals to the AFCS. Mechanical feedback is provided by the differentiating lever, as in the manual mode. When operating in the series mode, control surface displacement is not reflected at the control stick.

PARALLEL MODE.—In this mode, stabilizer movement is controlled by input signals from the AFCS alone. Both series and parallel mode solenoid valves are energized. Flight system pressure is ported to the electrohydraulic servo valve and the mechanical input lockout piston. Fluid pressure stabilizes the lockout piston and holds the mechanical input lever.

The transducer mounted on the servo ram provides an electrical signal feedback to the AFCS. There is no mechanical feedback, since the mechanical input is locked. Additional electrical signal feedback is provided by a transducer, which is mechanically linked to the stabilizer actuating arm. In the parallel mode, the control stick follows the motion of the stabilizer. Should the pilot desire to override the AFCS, he/she can overpower the lockout actuator with a stick force of 24 pounds.

Stop bolts are attached to the control stick pedal to limit fore-and-aft stick movement. The eddy current damper dampens out any rapid fore-and-aft stick movement.

All joints between the pushrods and bell cranks or idlers contain self-aligning bearings to compensate for any misalignment during operation and airframe deflections in flight that might cause binding.

Artificial feel is provided by the artificial-feel bungee. The bungee consists of two springs, which have different spring constants. The stick force caused by the bungee is proportional to stick displacement. At near neutral, the bungee provides a high stick force that decreases a short distance from neutral and gradually increases with the amount of stick displacement.

The electric trim actuator is mechanically linked to the artificial-feel bungee, and varies the neutral position of the bungee to provide longitudinal trim of the aircraft. The actuator consists of one high-speed and one low-speed motor, a gearbox, a brake, a ball detent clutch, and a threaded power screw. The actuator is manually controlled through inputs from the trim switch on the control stick grip. When the stabilizer is in automatic trim, the actuator receives inputs from the AFCS. High speed is used during manual trim, and low speed during automatic trim.

The stabilizer shifting mechanism, shown in figure 9-7, consists of a shift sector and its linkage, plus cable that runs from the flap drive gearbox and the rudder cam shift mechanism. A spin recovery cylinder is also attached to the shifting mechanism, and provides an alternate method of shifting the stabilizer and rudder from the "clean" configuration to the "dirty," or increased throw configuration.

In normal operation, when flaps are extended, a cable running from a drum on top of the flap drive gearbox to the sector assembly of the shifting mechanism rotates the sector. Linkage connecting the sector assembly and the control stick linkage is shifted. Linkage shifting increases control stick travel. Stabilizer down travel is increased to a 24-degree maximum. A cable is also connected from the sector assembly to the rudder cam stop shifting mechanism, which increases rudder travel from 4 to 35 degrees each side of neutral.

The pilot, at his/her option, may obtain increased stabilizer and rudder throw by actuation of the spin recovery assist switch, eliminating the necessity of lowering the flaps. This action ports hydraulic pressure through the spin recovery selector valve and its flow regulators and check valve to the spin recovery cylinder, causing it to extend and shift the mechanism in the same manner as provided by the cable action.

The two nonbypass-type filters in the system protect the intricate valving mechanisms of the actuator from contamination, and are vitally important to proper stabilizer operation. They are checked with the requirements listed in the maintenance requirements card deck, and should not be overlooked when troubleshooting stabilizer system malfunctions.

The stabilizer power package, used on various Navy aircraft, is linked to the approach power compensator system (APC). This system aids the pilot in maintaining optimum angle of attack for approach and landing. An APC potentiometer is mechanically linked to the power package, and provides electrical inputs to the APC system to compensate for changes in pitch attitude required during landing approaches. The APC system regulates the throttle position to provide the engine thrust required to establish and maintain the desired angle of attack. The potentiometer provides inputs relative to the position of the horizontal stabilizer.

Horizontal Stabilizer Control System (Double Axis)

Because of the complexity and interrelationships of the flight control systems of newer model aircraft, only a brief description of a representative stabilizer control (pitch/roll. axis) follows. This system allows pitch about the aircraft's lateral axis and roll about the aircraft's longitudinal axis.

Stabilizer control, which affects both the pitch and roll axis, is provided by forward or aft and/or left or right movement of the control stick grip. Forward or aft movement provides pitch-axis control; left or right movement, roll-axis control. The control, stick grip movement is mechanically transferred to the left and right stabilizer servo cylinders through the pitch and roll command summing network, the feel assemblies, and the summing network. These servo cylinders, which are normally powered by the flight and combined hydraulic power systems, move the stabilizers. If both hydraulic systems fail, the stabilizer servo cylinders automatically receive hydraulic power from the backup system. The trim switch on the control stick grip enables trimming of the aircraft in pitch and roll.

LATERAL CONTROL SYSTEMS

Lateral control systems control roll about the longitudinal axis of the aircraft. In this section, several of the different system arrangements used by aircraft manufacturers are described, and general maintenance requirements for primary flight control systems are discussed.

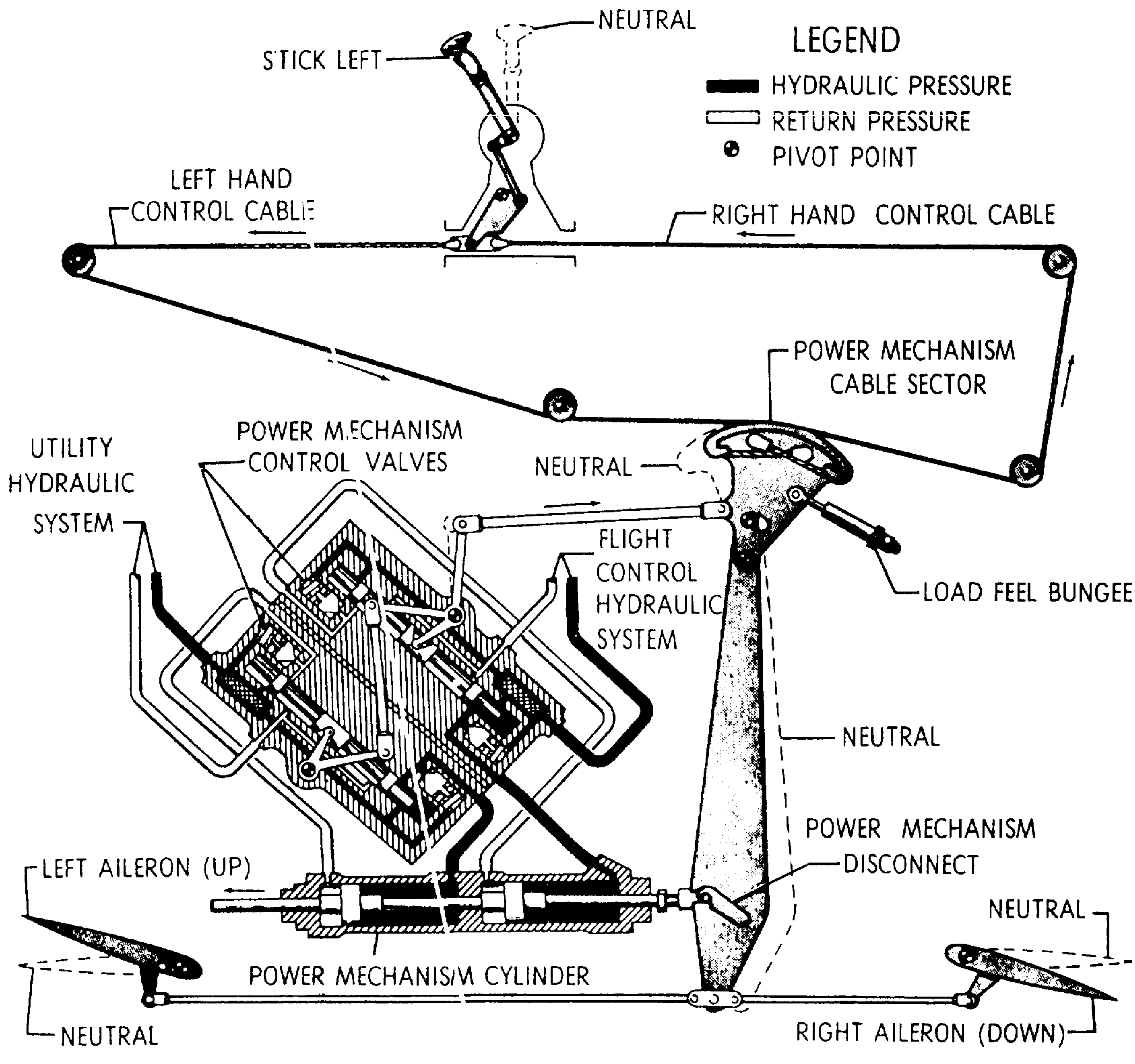


Figure 9-8.—Hydraulically powered aileron control system.

Aileron Control System

The aileron control system, shown in figure 9-8, is equipped with a power mechanism that provides hydraulic power to operate the ailerons. If hydraulic power fails, the mechanism can be disconnected, placing the system in complete manual operation. Movement of the aileron control system begins when the control stick in the cockpit moves left or right. When the stick is moved, cables connected to the bell crank in the control stick housing are moved to operate the sector on the power mechanism. With the actuation of the sector, the power mechanism operates, transferring the movement to the mechanical linkage that operates the ailerons.

The aileron power mechanism consists of two control valves, a dual-chambered hydraulic power cylinder, cable sectors, and a system of latches and related cranks. Linkage connects the control valves in tandem. The flight control hydraulic system powers one valve, and the other is powered by the utility hydraulic system.

The power cylinder is a single tandem cylinder, composed of four chambers with pistons connected to a common shaft. Each of the two control valves operates on that portion of the power cylinder to which it is associated. Both hydraulic systems operate simultaneously, and each delivers 3,000-psi pressure to the mechanism. If one hydraulic system should fail, the other system will supply enough power to operate the ailerons at reduced hinge movement.

When the control stick moves, the control cables move the power mechanism sector. Through linkage, the sector operates the control valves, which direct hydraulic fluid to the power cylinder. The cylinder actuating shaft, which is connected to the power crank through a latch mechanism, operates the power crank. The crank moves the push-pull tubes, which actuate the ailerons. In the event of complete hydraulic power failure, a handle in the cockpit maybe pulled to disconnect the latch mechanisms from the cylinder. When the handle is pulled, it places this particular aileron system in complete manual operation. In manual operation, the power cylinder is disconnected from the cable sector, causing the control stick to manually move the ailerons at a reduced rate.

The lateral control system incorporates a load-feel bungee, which serves a dual purpose. See figure 9-9. The bungee provides an artificial feel and centering device for the aileron system. It is interconnected between the aileron system and the aileron trim system. Energizing the aileron trim actuator moves the bungee operating the power mechanism, which repositions the aileron control system to a new neutral position.

In normal operation of the control system, when the control stick is actuated left or right, the power mechanism compresses the bungee. The compressed bungee returns the stick to the neutral position upon release of the stick.

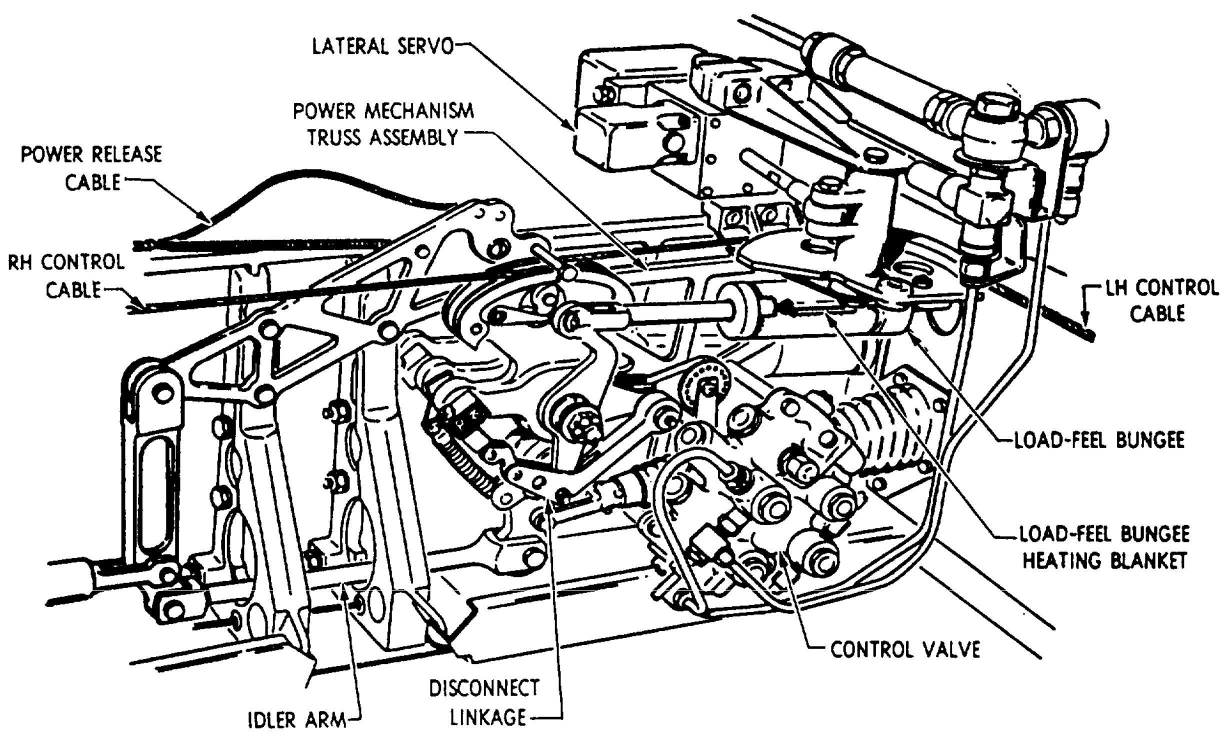
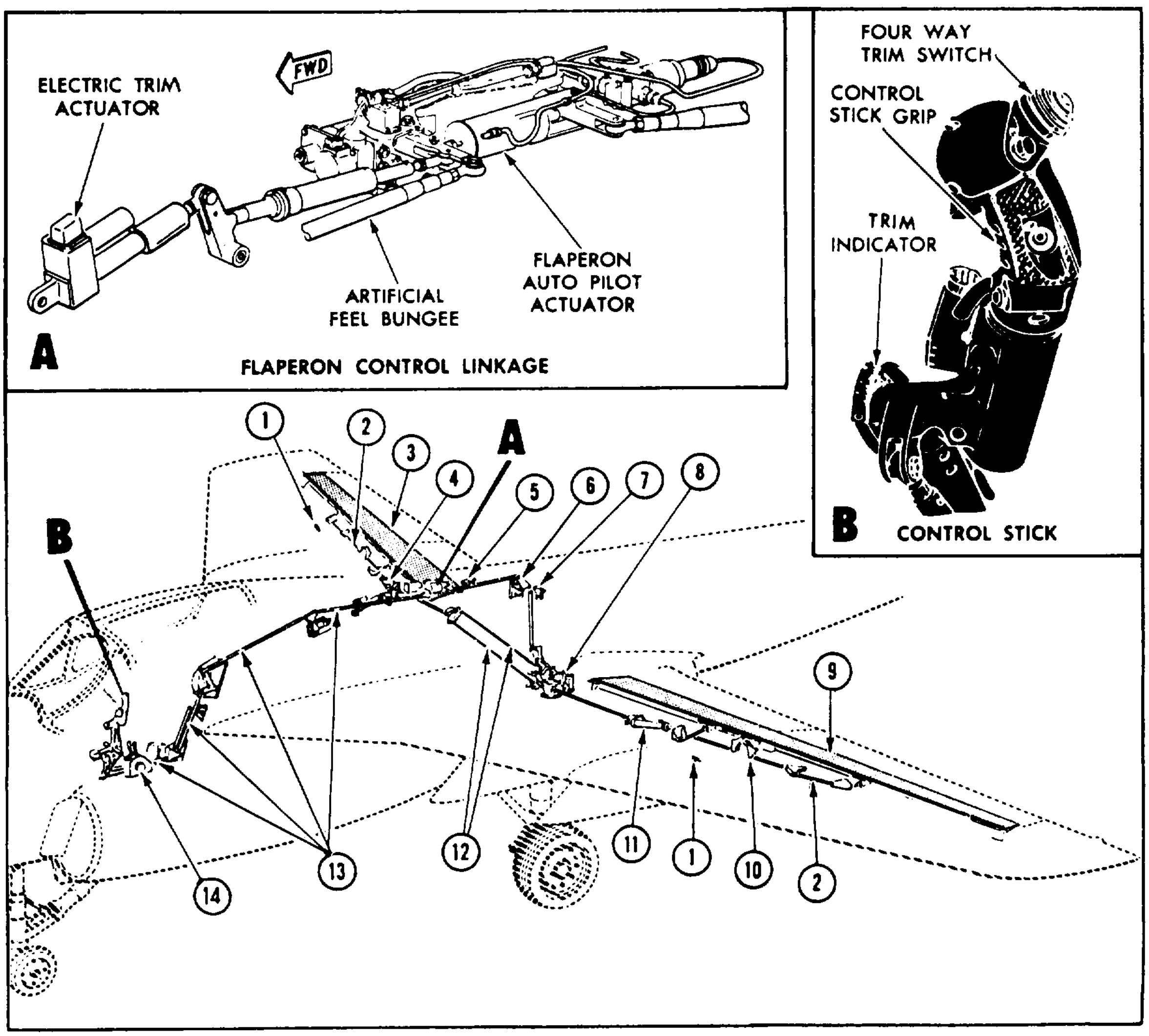


Figure 9-9.—Aileron power mechanism.

Flaperon Control System

The flaperon control system, shown in figure 9-10, is an example of lateral control provided by an electrohydraulic-mechanical flaperon system. The system includes an inboard and outboard flaperon for each wing and three actuators (a single flaperon autopilot actuator and a flaperon power actuator in each wing).

Control stick movement, left or right, raises the respective two flaperons, while the opposite two remain flush with the wing. Full throw of the control stick by the pilot causes the inboard flaperon to rise 49 1/2 degrees and the outboard flaperon to rise 53 degrees. In flight, the flaperon can also be positioned by the AFCS. Control stick movements are transferred through the pushrod and bell crank system to the flaperon autopilot actuator. Mechanical outputs from this actuator are conveyed to a gearing mechanism, at which point linkage to the left and right wing flaperon power actuators separates. The gearing mechanism transmits movement to the left or right flaperon, while the opposite flaperon is



1. Wing-fold flaperon interlock switch
2. Flaperon control linkage
3. Right wing flaperons
4. Flaperon actuator (right wing)
5. Flaperon pop-up valve
6. Wing-fold interlock mechanism
7. Filter
8. Flaperon pop-up mechanism and cylinder
9. Left wing flaperons
10. Flaperon control linkage
11. Flaperon actuator (left wing)
12. Crossover cables
13. Pushrods
14. Throttle quadrant

Figure 9-10.—Flaperon control system.

maintained flush with the wing. When the flaperon pop-up cylinder is actuated, the gearing mechanism transmits pop-up motion to each wing flaperon power actuator.

The semiautomatic flaperon pop-up device aids in reducing ground roll during landing. The pop-up system is activated by the pilot placing the flaperon pop-up switch in the ARM position. All flaperons (four) will then automatically pop up approximately 41 degrees when the aircraft weight is on the landing gear and the throttles are retarded.

A mechanical interlock device prevents damage to the flaperons during folding of the wings. When the wings are folding, the flaperons cannot be extended. In addition, the folding operation cannot start unless the flaperons are flush with the wings.

A wing-fold interlock prevents flaperon pop-up after the wings are folded. A fail-safe spring returns the flaperons to the flush position in case the combined hydraulic system or electrical system should fail.

The eddy current damper links mechanically to a bell crank in the flaperon control linkage. See figure 9-11. It dampens any rapid left or right control stick movement by producing an opposing force proportional to the speed at which the stick is moved. The damper contains permanent magnets, a rotating copper disc, a gear train, and a clutch assembly. Control stick motion rotates the clutch and gear train, which, in turn, rotates the copper disc. The copper disc is sandwiched in the air gap between the six permanent magnets and a flux plate. As the copper disc revolves, the magnetic field between the magnets and the flux plate is disturbed, causing an opposing force (eddy currents) that tries to stop the disc. The opposing force is proportional to the speed of the rotating disc and to the speed of stick movement. The clutch will slip at a force of 275 to 325 inch-pounds to prevent control stick binding if the damper jams.

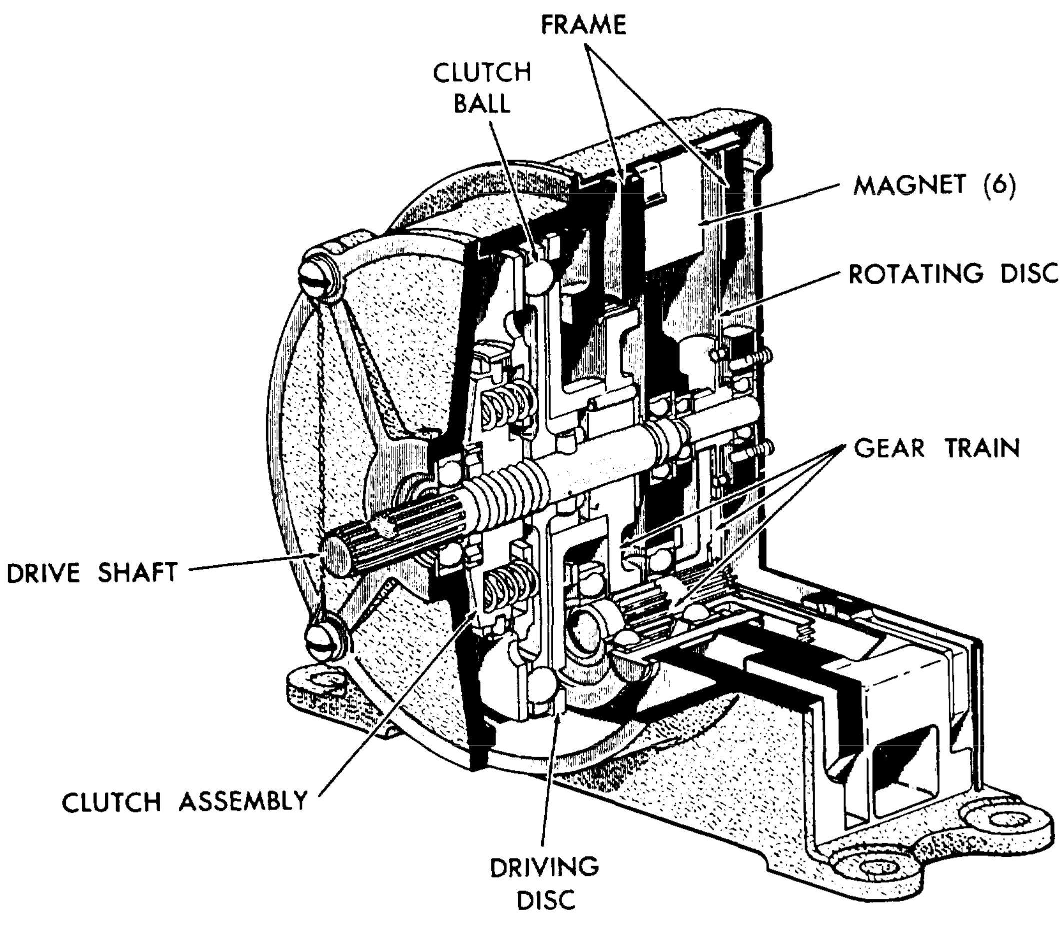


Figure 9-11.—Eddy current damper.

Figure 9-12 illustrates a representative flaperon control system. The flaperon autopilot actuator is powered by the flight hydraulic system and transmits mechanical movement to the flaperon power actuators. The flaperon power actuators are tandem type and powered by the combined and flight hydraulic systems. They are capable of operating on only one system if one system should fail.

The artificial-feel bungee provides an initial control stick preload and increased force feel over the full range of stick displacement. The electro-mechanical actuator provides lateral trim, which varies the neutral position of the artificial-feel bungee. Trim is set by the switch on the control stick grip. The pilot may read the mechanical flaperon trim indicator on the control stick. See figure 9-10.

AUTOPILOT ACTUATOR.—The flaperon autopilot actuator (figs. 9-12 and 9-13) contains an electrohydraulic servo valve, actuator pistons, solenoid valve, transducer, series link, and series-link rod. It indirectly controls flaperon movement in response to mechanical movements from the pilot. It receives electrical inputs from the automatic flight control system. The actuator can operate in two modes—manual or series.

In manual mode, the solenoid valve is de-energized and no fluid is ported to any part of the actuator. The actuator piston rod is free to idle. The series-link cylinder acts as a rigid link that transfers input lever motion to the output lever.

In series mode, the solenoid valve energizes and ports pressure to the servo valve. Pressure from the servo valve drives the actuator pistons together. This pressure causes the pistons and the rod to act as one piece. When the servo valve is at null, pressures in the piston end chambers are equal. Electrical signals

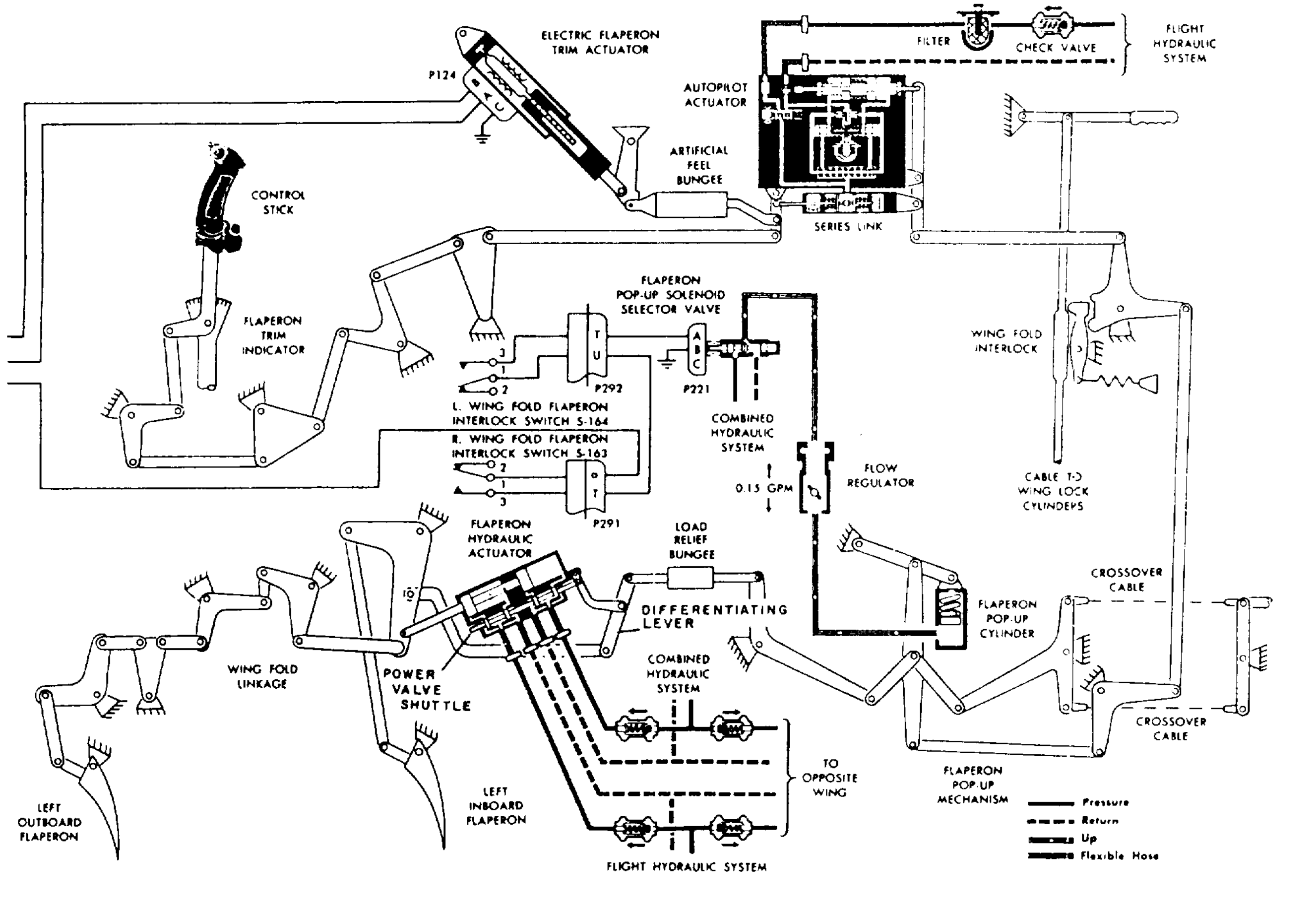


Figure 9-12.—Flaperon control system.

from the automatic flight control system cause the electrohydraulic servo valve to differ the pressures in the end chambers. The signal provides the working force for the actuator. The actuator piston rod drives the output lever. Pressure at the series link compresses a lock spring, unlocking the series link. The actuator can stroke the pilot-commanded piston. When the pilot moves the input link, relative motion between input and output causes the transducer to send a signal to the AFCS amplifier. The signal combines with other flight stability signals, and the resultant signal operates the servo valve. The AFCS can be overridden by the pilot applying a stick force of 25 pounds.

SYSTEM ACTUATORS—The flaperon system actuators directly control the flalperon movement in response to mechanical movement from the autopilot actuator. The actuator (fig. 9-1 2) consists of two tandem-mounted power pistons and a power valve shuttle. Mechanical inputs are introduced through the load-relief (safety) bungee and the valve input lever to the power valve shuttle portion of the actuator. The inputs cause a valve error and the porting of hydraulic pressure to the power pistons. As the flaperon moves, mechanical linkage attached to the actuator tends to null this valve error, The power valve shuttle returns to neutral. The flaperons remain in the selected position until new mechanical inputs are received from the pilot or the AFCS.

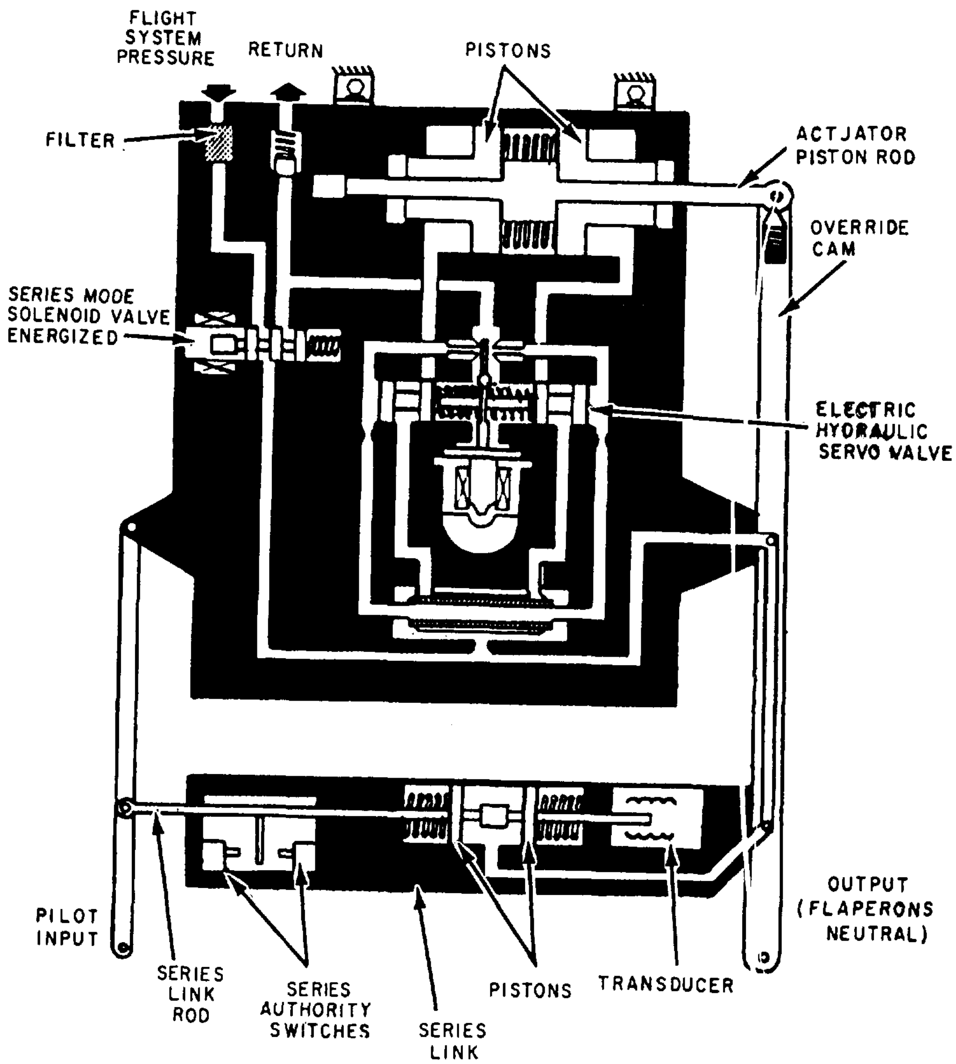


Figure 9-13.—Flaperon autopilot actuator.

Combination Aileron/Spoiler Deflector System

Navy aircraft employ more than one system for lateral control of the aircraft. Figure 9-14 shows an aileron and spoiler/deflector arrangement to achieve an increased roll rate about the longitudinal axis.

In this system, left and right control stick movements transfer mechanically to the aileron and spoiler/deflector control linkage. The viscous damper cylinder is connected in the linkage. It resists rapid control stick movement, presenting overcontrol of the aileron system when the control augmentation mode of the AFCS is engaged. The control augmentation mode of the AFCS improves lateral and longitudinal stability of the aircraft.

The load-limiting links located throughout the system protect control linkage and components from excessive loads These links have a breakout force, so they normally act as a fixed link. Loads that exceed the breakout force cause the links to extend or retract and absorb the overload.

Artificial feel is provided by the mechanical feel spring assemb]y. The assembly simulates air load resistance at the control stick. When released, the control stick returns to neutral by the feel spring preload.

The roll-feel isolation actuator prevents excessive forces from reaching the control stick. When the control stick is deflected, linkage to the feel isolation actuator servo valve repositions the servo valve slider and directs hydraulic pressure to the actuating pistons. The cylinder housing is connected to the control linkage and moves in the direction corresponding to stick movement. As the cylinder housing moves, the servo valve slider repositions to neutral, blocking fluid flow to and from the actuator until new inputs are initiated.

The AFCS roll actuator connects to the control linkage by a scissor link. Normally, this scissor link acts as a simple idler. When the actuator receives signals from the AFCS, it causes, the linkage to act as a variable link. This action produces control system inputs completely independent of the control stick.

Output motion from the AFCS linkage is transmitted through control system linkage to the aileron trim and mixing linkage. The mixing linkage directs inputs to both the aileron and spoiler/deflector linkage. Dead-band stops within the mixing linkage allow the ailerons to reach a trailing edge up position of 2 degrees 30 minutes, 15 minutes, before any spoiler/deflector motion is initiated.

The power control cylinders for the ailerons and the spoiler/deflectors are tandem type. Power control No. 1 and power control No. 2 hydraulic systems supply hydraulic pressure. Half of the servo valve on each cylinder directs PC No. 1 hydraulic pressure to the corresponding half of the PC cylinder. The second half of the servo valve directs PC No. 2 hydraulic pressure to the other half of the cylinder. If one system fails, the other system operates the ailerons and spoiler/deflectors.

Input control linkage connected to the servo valve control arm of the PC cylinders positions the valve slider to direct pressure to the actuating pistons. The actuating piston extends or retracts the cylinder housing. As the cylinder housing moves, the servo valve control arm repositions the servo valve slider. When the ailerons and spoiler/deflectors position is equal to the demand input, the servo valve slider is again at neutral. Fluid flow is blocked to and from the cylinder until a new control system input is initiated.

The spoiler/deflector on each wing operates with the upward throw of the aileron on that wing. They are located in the left- and right-hand wing center sections, forward of the flaps. The spoiler extends upward into the airstream, disrupting the airflow and causing decreased lift on that wing. The deflector extends down into the airstream and scoops airflow over the wing surface aft of the spoiler, preventing airflow separation in that area.

A stop bolt on the spoiler/deflector bell crank limits movement of the spoiler to 60 degrees of deflection. The deflector is mechanically slaved to the spoiler. It can be deflected to a maximum of 30 degrees when the spoiler is at 60 degrees. The spoiler deflectors open only with the upward movement of the ailerons. They are normally closed. The linkage motion lost when the aileron is down is absorbed by the spoiler deflector load-limiting link.

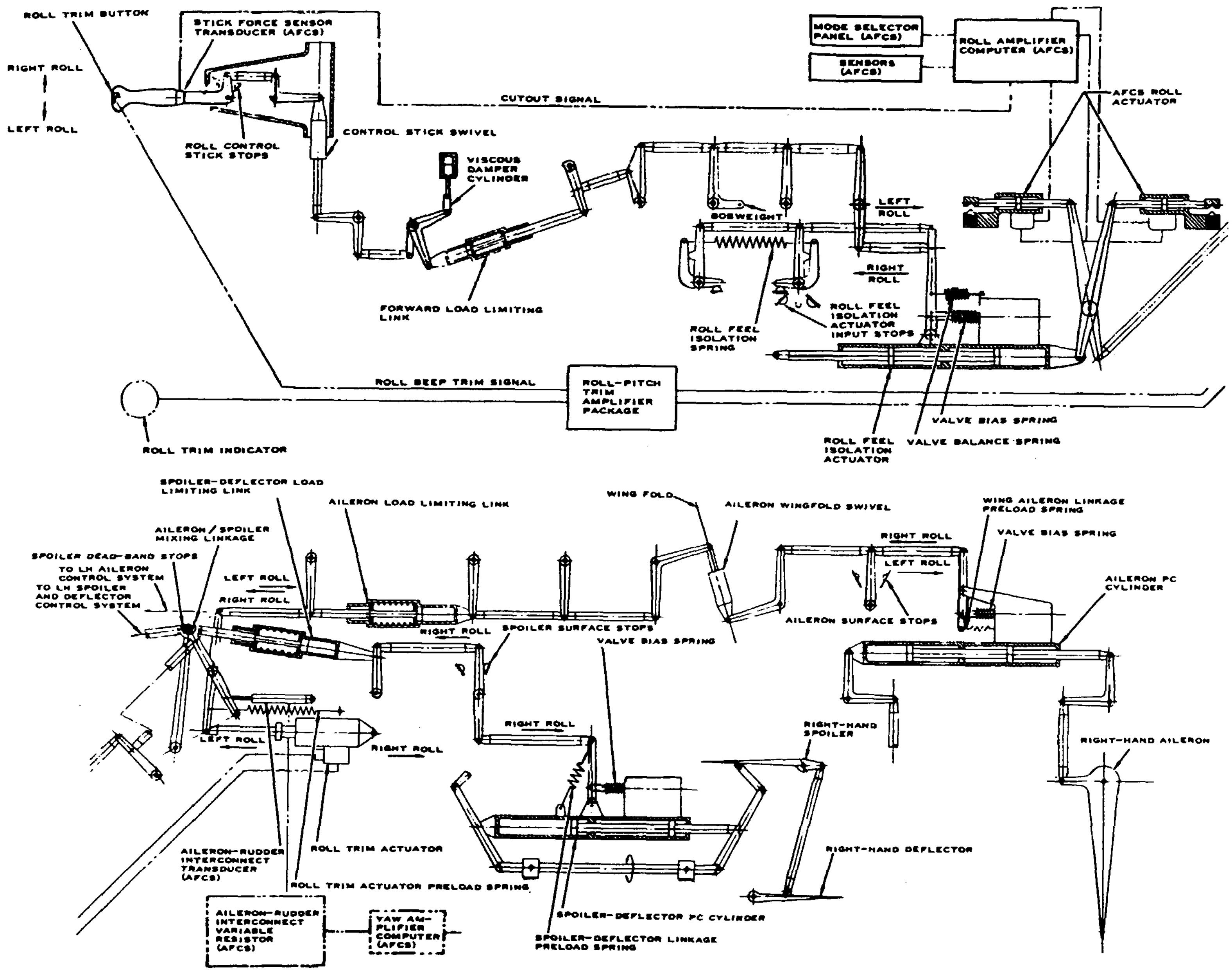


Figure 9-14.—Aileron and spoiler/deflector system.

Spoiler Control System

On one model aircraft, spoiler action is provided through the control stick grip, roll command transducer, roll computer, pitch computer, and eight spoiler actuators (one per spoiler). When used to increase the effect of roll-axis control, the spoilers can only be controlled when the wings are swept forward at 57 degrees.

Right or left movement of the control stick grip mechanically transfers to the roll command transducer. The transducer converts the movement to inboard and outboard spoiler roll commands. Because the spoilers are vital for landing, the left- and right-wing inboard and No. 1 mid-spoilers are controlled by the roll computer. The spoilers are powered by the combined hydraulic power systems. The left and right outboard and No. 2 mid-spoilers are controlled by the pitch computer. These spoilers are powered by the mid-outboard spoiler/high lift backup module. This combination provides positive spoiler control if either computer or hydraulic power source malfunctions.

DIRECTIONAL CONTROL SYSTEMS

Directional control systems provide a means of controlling and/or stabilizing the aircraft about its vertical axis. Most Navy aircraft use conventional-type rudder control systems for this purpose.

The rudder control system, shown in figure 9-15, is operated by the rudder pedals in the cockpit. The system is powered hydraulically through the rudder actuator. In the event of hydraulic power failure, the hydraulic portion of the power system is bypassed. The system is then powered mechanically through control cables and linkage. An aerodynamic irreversible hydraulic system is employed in the rudder system. To accomplish trim, the complete rudder surface is repositioned.

The actuation of the rudder pedals causes the control cables to move a cable sector assembly. The cable sector, through a push-pull tube and linkage, actuates the power mechanism. The rudder actuator deflects the rudder to the left or to the right.

A load-feel bungee is connected to the push-pull tube, and is compressed when the push-pull tube is actuated. When the pedals are released, the compressed bungee returns the system to the neutral position. In the event of hydraulic failure, a slip link allows movement of the control valve linkage to port hydraulic fluid from the actuating cylinder. Then the cylinder can be mechanically driven by pilot input during manual operation. In manual operation, surface travels are reduced by the lost-motion effect of the slip link.

The load-feel bungee is also the connecting link from the rudder trim actuator to the power mechanism. When the trim actuator is operated, the bungee repositions the power mechanism. The power mechanisms deflect the rudder for nose-left and nose-right trim. Figure 9-15 is a functional schematic of the operation of the rudder control system.

The rudder power mechanism is actuated when movement from the cable sector assembly is transmitted through the push-pull tube to the primary control crank. The crank is connected to the load-feel bungee, a slip link to the secondary crank, a link and spring to the pedal position transmitter, and a link to the control valve of the actuator assembly. The actuator assembly consists of an electromechanical dual input control valve, a rudder surface position transmitter, and a power cylinder. When the mechanism linkage is actuated, the control valve directs hydraulic pressure from both the utility hydraulic system and the surface control hydraulic system to the power cylinder. The valve directs the hydraulic pressure to two separate chambers in the cylinder. Each chamber has a separate piston that is mounted on as common shaft. The shaft is connected to a push-pull tube that moves the rudder. The actuator assembly normally operates from both hydraulic systems. If one system should fail, the other supplies sufficient pressure to operate the rudder with some lost hinge movement. In the event both hydraulic systems fail, the slip link will allow movement of the control valve linkage to port hydraulic fluid from the actuating cylinder.

When the automatic flight control system is engaged, the actuator initiates the movement of the rudder system through the electrical impulse received by the control valve from the surface control amplifier. The pedal position transmitter and the rudder surface transmitter function only when the automatic flight control system is engaged.

Rudder pedal movement transfers mechanically to the left and right rudder servo cylinders through the rudder feel assembly, the yaw summing network, and the reversing network. These servo cylinders, normally powered by the flight and combined hydraulic power systems, move the rudders. If both hydraulic systems fail, the rudder servo cylinders automatically receive hydraulic power from the backup hydraulic system (flight control backup module). The rudder trim switch on the EXT ENVIRONMENT/THROTTLE control panel enables trimming of the aircraft in yaw. Setting the switch to L or R provides a trim-left or trim-right input, respectively, to the rudder trim actuator. The actuator provides rudder movement through the rudder-feel assembly, the yaw summing network, the reversing network, and the rudder servo cylinders.

ELECTRONIC CONTROL SYSTEMS

All electronic flight control servo cylinders are controlled by electrical impulses from computers. The computers compare all data received from the pilot's control stick, airspeed indicator, altimeter, angle of attack, and other sensors. They configure all flight controls for best flight characteristics and performance of the aircraft. An example of the electrical portion that replaces the mechanical

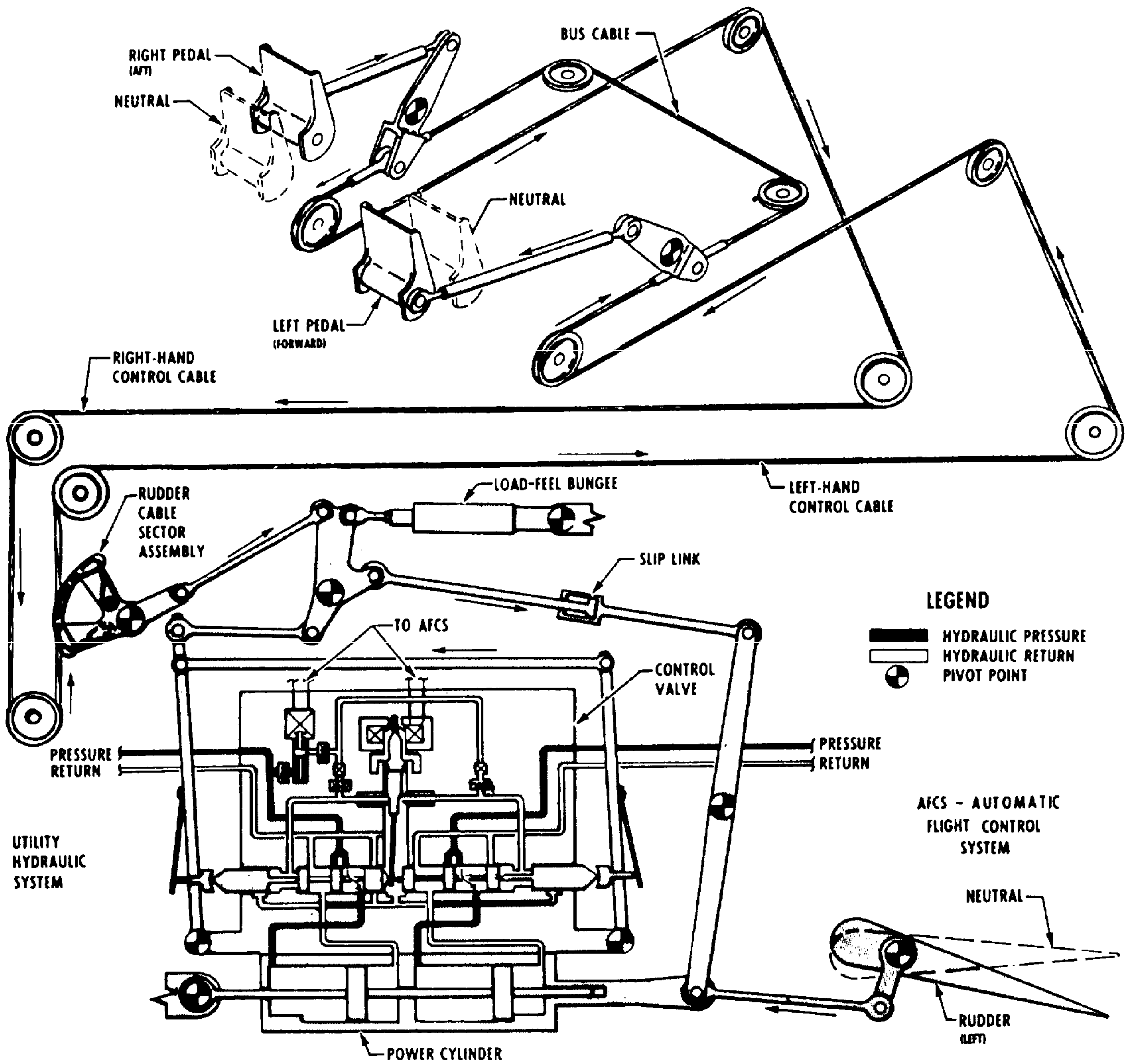


Figure 9-15.—Rudder control system.

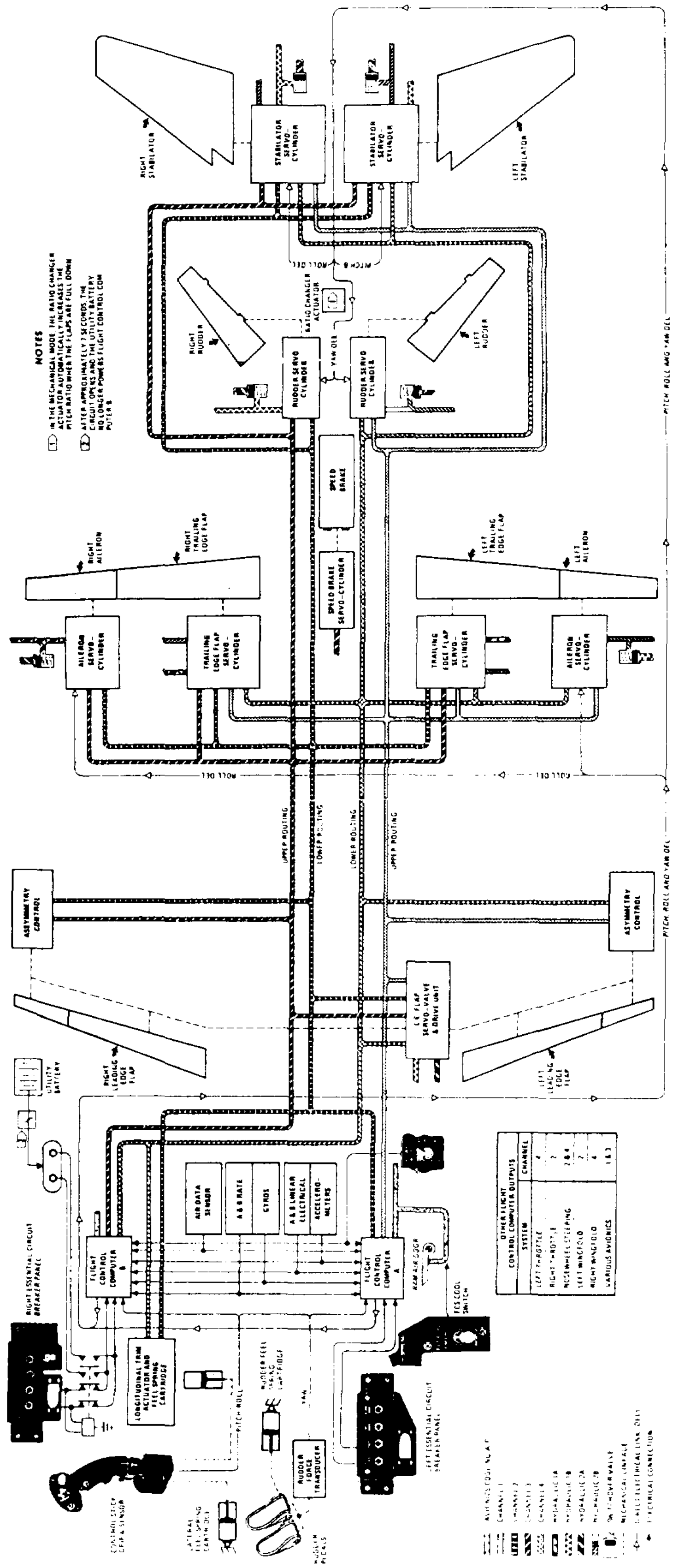


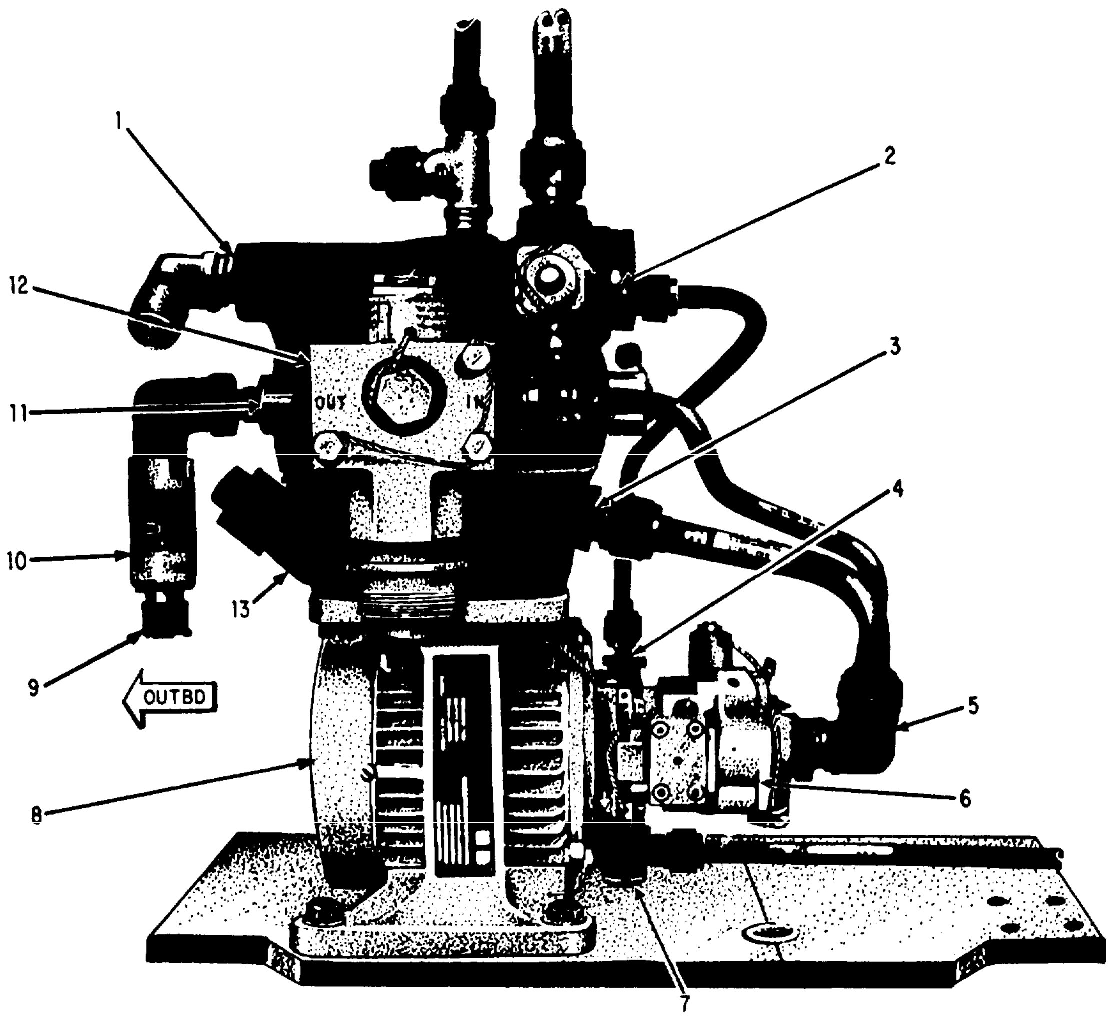
Figure 9-16.—Electronic flight controls.

linkages system is shown in figure 9-16. Each electric component is duplicated two to four times throughout the system. Provisions are made to detect a failed component or sensor and remove its influence from the system. These multiple redundant paths ensure that a single failure has no effect, and multiple failures have minimum effect on controls.

BACKUP SYSTEM

Despite the dual system design requirement for flight control systems, a complete hydraulic system failure is possible. System failure could be a result of component or plumbing failure or as a result of enemy-inflicted damage. The backup flight control system, shown in figure 9-17, provides for an additional measure of flight control safety. The system activates whenever a partial or complete hydraulic system failure occurs.

The complete backup flight control system is mounted on a protective armor plating that measures only 8 by 16 inches and is located close to the rudder and stabilizer power packages. Flight and combined



1. Rudder return port
2. Case drain (reservoir)
3. Suction port
4. Case drain (pump)
5. Suction and pressure ports
6. Pump
7. Seal drain port
8. Motor
9. Electrical receptacle
10. Pressure switch
11. Check valve
12. Filter assembly
13. Reservoir assembly

Figure 9-17.—Backup flight control hydraulic system.

hydraulic system pressure line switches control the operation of this system.

The two switches in the pressure lines to the backup flight system are wired normally closed at zero pressure. The backup pump outlet pressure switch is wired to normally open at zero pressure. The switches actuate at 900-1,100 psi on rising pressure and 700-900 psi on decreasing pressure. Closing of the combined or flight system pressure switches energizes the backup system motor pump. Closing the outlet pressure switch lights the backup hydraulic system indicator light on the annunciator panel in the cockpit.

When pressure in the flight and/or combined hydraulic system decreases to 700-900 psi, the system is automatically activated. The system isolates a portion of the combined system in the tail of the aircraft by check valves in the pressure lines and a shutoff valve in the return line. When the shutoff valve closes, it stores a full charge of fluid in the backup system reservoir,

The reservoir mounts on top of the motor-pump assembly. It has a capacity of 0.84 quarts. The return system shutoff valve is an integral part of the reservoir end flange inside the reservoir pressurizing spring. The soft-seated, poppet-type shutoff valve is held open when the reservoir is at the full position. When pressure drops and the reservoir piston moves about three-sixteenths of an inch away from the full position, the spring-loaded valve closes and prevents flow from the reservoir. The shutoff valve also acts independently as a relief valve to relieve reservoir pressure above 95 psi.

Return fluid flow from the rudder and stabilizer actuators fills the backup system reservoir. When the reservoir approaches the full position, it mechanically opens a shutoff valve, allowing return flow to go to the combined system reservoir. In normal flight, the 40-psi return system pressure is enough to maintain the backup reservoir piston at the full position. The shutoff valve fully opens against its spring pressure.

If return system pressure drops below the reservoir pressurizing spring pressure of 15 psi, the reservoir piston moves and displaces fluid through the shutoff valve. As the piston moves, the shutoff valve closes fully in three-sixteenths of an inch of piston movement.

The shutoff valve may open momentarily during backup system operation to discharge excess fluid volume. This action may be a result of unequal stabilizer in-and-out stroke volume or thermal expansion of the fluid. The shutoff valve also opens when the flow rate exceeds the flow capacity of the backup pump. The latter condition could occur when the flight system is operating normally and high rate inputs are applied to the actuators.

Pressure line isolation is accomplished by the use of check valves. To prevent backup system leakage to a failed combined system, a soft-seat check valve is installed upstream of the standard metal-seat check valve. These valves are found in the combined system pressure line.

A three-position backup system hydraulic test switch is located in the cockpit. The central spring-loaded OFF position provides automatic function in flight. The momentary hold positions, COMBINED and FLIGHT, are for a ground test of the system when the aircraft is on external electrical power. Selection of either position will energize the motor pump when aircraft pressure is less than 700-900 psi.

A cartridge-type filter element housed within the reservoir head and a pressure line filter protects the system from contamination. Since the backup motor pump is energized when either or both primary systems fail, the following three operational conditions can exist:

1. With the backup and flight systems operating, normal flight control is available. The backup system performs as an isolated system with the return shutoff valve closed. The variable displacement backup motor pump has a maximum rated output of 3 gpm at 1,000-psi output pressure to zero gpm at cutoff pressure (3,000-3,200 psi). The pump cannot match the high rate capacity of the flight system. Backup motor pump pressure will drop to zero when demand exceeds 3 gpm. Zero pressure causes the cockpit indicator light to go out. When pressure increases to 900-1,000 psi, the light will come on again, indicating backup system operation.

2. With the backup system and combined system operating, normal flight control is available. The backup system is not isolated, as normal combined system pressure exists within pressure and return lines. The return shutoff valve remains open. Combined system pumps maintain high pressure at the rudder and stabilizer actuators. Flow demand on the backup pump is not excessive at high rates. The cockpit indicator light should remain on, indicating backup system operation.

3. When the flight and combined systems fail, the backup flight control system performs as an isolated system. Surface rates available at the rudder and stabilizers are reduced by the limited output of the backup pump. There is no flaperon actuator control. The cockpit indicator will flicker out if the pilot applies inputs to the controls that exceed the capacity of the pump. The cockpit RUDDER THROW light will also be illuminated, indicating that approximately 33 percent of normal rudder throw is available.

POWER ACTUATOR MAINTENANCE

Maintenance of primary flight control surface power actuators is generally beyond the capability of organizational maintenance-level activities. Removal of hydraulic components and associated linkages on the power actuators will destroy critical adjustments. Readjustment requires special tooling, jigs, and other equipment available only at intermediate- or depot-level maintenance facilities. When a power mechanism has been isolated as the cause for flight system malfunction, it is removed. It is forwarded with the accompanying paperwork to the supply activity for disposition.

CONTROL SYSTEMS MALFUNCTIONS

There have been many cases reported in which, after flight, pilots have found flight controls jammed while the aircraft was on the ground. Because the controls were freed by excessive pressure before an inspection could be made, the causes for the jammed condition could not be found. No positive corrective action was taken before the aircraft were released for flight. In some cases, accidents occurred on such aircraft shortly thereafter.

When an aircraft experiences a control discrepancy during flight, a thorough investigation should be conducted immediately. In cases where aircraft have safely returned from a flight during which a control discrepancy was experienced, a thorough investigation is necessary. This investigation must be made before further flight. All parts of the affected control system should be inspected for proper rigging, clearances, and potential causes for interferences. All sealed units that are suspect must be replaced. Primary cause factors that should not be overlooked include maneuvers that have exceeded the operational design of the control systems. Hydraulic system contamination, corrosion and/or distorted or disconnected linkage may have caused the problem. Inadequate lubrication and external contamination in the form of preservative compounds, such as grease combined with dirt and dust, may have caused the problem. An increasing number of flight control system malfunctions are related to system contamination, and this ever-important aspect of hydraulic system maintenance should be given the attention it deserves. Checking of system filters and contamination inspection of suspected systems are within the capability of organizational activities. If a system is found to be contaminated, the source of contamination must be eliminated and the system cleaned by recycling or flushing in accordance with instructions provided in the appropriate MIM. Contaminated components must be replaced as necessary to restore proper system operation.

Disposition instructions for removed hydraulic components vary with the production status of the aircraft model. Diligent care must be taken to retain the component in the *as-is* condition, with no change in adjustment, disassembly, or cleaning. If the component has slides or pistons that are jammed, no attempt to free them should be made.

The aircraft must not be released for further flight until the cause has been determined and corrected. If it is not readily apparent why the component malfunctioned, you should submit a Hazardous Material Report/Engineering Investigation request. If the discrepancy cannot be duplicated or cause determined, an appropriate entry must be made in the Miscellaneous History section of the aircraft logbook.

TROUBLE ANALYSIS

Trouble analysis of the flight control systems requires the same systematic approach as any other hydraulic system. In many instances, malfunctions are written off with incorrect corrective actions on the maintenance action form (MAF). The corrective action, Could Not Duplicate, or Replaced Suspected Component, often results in a repeat discrepancy or loss of the aircraft. Thoroughness in determining the cause of a malfunction cannot be overemphasized.

Trouble analysis of the flight controls will require complete cooperation with other work centers that are involved in the operational checkouts. Most flight control systems have electrical input, as well as mechanical input from autopilot, automatic flight control systems, or stabilizing augmentation systems. Inputs occasionally cause erratic and/or misleading aircraft flight characteristics. Flight characteristics can be misinterpreted, and the resultant write-up in the aircraft discrepancy portion of the aircraft flight record book may be vague or misleading. To gain further insight regarding the vague discrepancy, the maintenance crew should question the pilot who experienced the malfunction.

Isolating the mechanical and hydraulic portion of the flight control system from systems that provide automatic input will serve to pinpoint the actual problem area. The MIM provides troubleshooting/ trouble analysis aids and appropriate schematics. The MIM allows for the systematic checking out of the system and associated components. In some MIMs these aids are general in nature and limited to the more common causes of failure. Several MIMs combine the operational checkout procedures with trouble analysis aids. Steps of the checkout procedures are performed in rigid sequence, and any discrepancy must be corrected before proceeding to the next step.

A thorough knowledge of the system involved and consistent use of the mechanical and hydraulic schematics will expedite the trouble analysis process. Excessive time required for troubleshooting should be documented on a separate VIDS/MAF. This will separate the actual repair time from troubleshooting time. Separate VIDS/MAFs provide more accurate input information to the Maintenance Data Reporting System.

When the malfunction has been determined and corrected, the complete system should be operationally tested. Testing should occur in all modes of operation to verify system integrity. Quality assurance inspection during repair progression, testing, and of the end product is a must. When prescribed in the applicable periodic maintenance information cards, test flight requirements are mandatory. The test flight pilot is briefed by a qualified quality assurance representative regarding the nature of the discrepancy and corrective action taken.

ALIGNMENT AND OPERATIONAL CHECKS

Procedures for rigging flight control systems vary with each type of aircraft. Applicable MIMs provide a list of tools, special equipment, preparatory considerations, and step-by-step instructions for rigging systems.

On some aircraft, the system rigging divides into a series of sections, such as the control stick, control mechanism, power control actuator, and cables. If only that section of the system has been affected, it may not be necessary to rig the complete system.

Pushrods, bell cranks, and idlers are installed so that end play is eliminated. They should be free to rotate without binding. Cables should be inspected for corrosion, broken strands, and proper tension. Correct cable tension is necessary to obtain proper response of the control surface. Low cable tension may cause sluggishness, free play, and flutter of the control surface. Excessively high cable tension will cause increased system friction and may result in damage to pulleys, bell cranks, or the cable itself.

A variety of fixtures, pins, and blocks are available for performing alignment and rigging checks on flight control systems. Neutralizing (locking the controls and linkage in a predetermined position), as described in the aircraft MIM, is required during the alignment and adjustment of the flight controls.

NOTE: Installation and removal of the fixtures, pins, and blocks should not require excessive force. Slight pressure is permissible because of the system tolerance and temperature effects on the aircraft. Always refer to the MIM for tolerance information.

Figure 9-18 shows the throw board used to check the travel of a horizontal stabilizer. The throwboard is held in place by two wingnut attachment screws. Before tightening these screws, the throwboard is positioned so that the alignment hole at the zero-degree mark is in line with the alignment screw in the aircraft fuselage.

Control surface throws may be measured in degrees and minutes or inches and fractions. Figure 9-19 provides an example of an aileron throw indication in degrees (°) and minutes ('). The protractor scale is calibrated in 30-minute increments. The indicator reads 3 degrees 40 minutes obtained as follows:

1. Read 3 degrees 30 minutes, as shown on the protractor scale.

2. Since the indication mark does not fall directly on the calibrated mark of the protractor scale, look for the closest alignment of indicator and protractor calibrated marks in the direction of indicator travel.

Read the value from the 0-minute mark on the indicator to the closest alignment, which, in this example, is 10 minutes.

3. Add 3 degrees 30 minutes and 10 minutes to get the true indication of 3 degrees 40 minutes up travel.

Each mode of operation that was affected by alignment or malfunction and subsequent repair action must be operationally checked, and the success of the checkouts verified by a qualified quality assurance representative. All maintenance, including alignment, adjustment, operational testing, and component replacement, must be in accordance with the instructions provided in the applicable MIM.

CABLE AND RIGID CONTROL SYSTEMS MAINTENANCE

Cable and rigid control systems maintenance includes inspection to discover actual and potential defects, servicing with lubricants, and correction of reported malfunctions and defects. Malfunctions that occur in control systems include frayed and loosened bearings, unnatural tightness (binding), and broken or damaged components.

Cable Control Systems

Cables have many advantages. They will not sever readily under sudden strains. Cables are stronger than steel rods or tubing of the same size. They flex without setting (permanent deformation) and can be led easily around obstacles by using pulleys. Cables can be installed over long distances (such as in large aircraft) without a great degree of sagging or bending. Vibration will not cause them to harden, crystallize, or break, as may be the case with push-pull control rods. Because of the great number of wires used in cables, cable failure is never abrupt, but is progressive over periods of extended use. When used for the manipulation of a unit in a control system, they are usually worked in pairs-one cable to move the unit in one direction, the other to move it in the opposite direction. Weight is saved in spite of a second cable because the push-pull rod needed to cause a similar movement in a unit would have to be quite thick and heavy (comparatively speaking). Since cables are used in pairs and are stretched taut, very little play is present in system controls, and no lost motion exists between the actuating device and the unit. Consequently, cable-controlled units respond quickly and accurately to cockpit control movement. In some simple cable systems, only one cable is used, and a spring provides the return action.

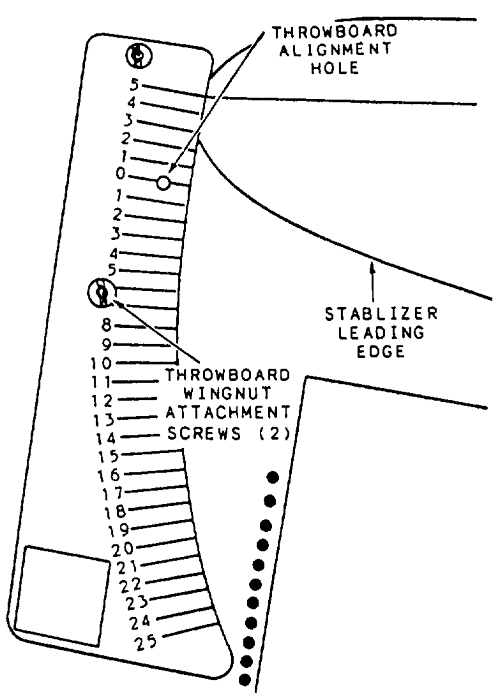


Figure 9-18.—Stabilizer throwboard installation.

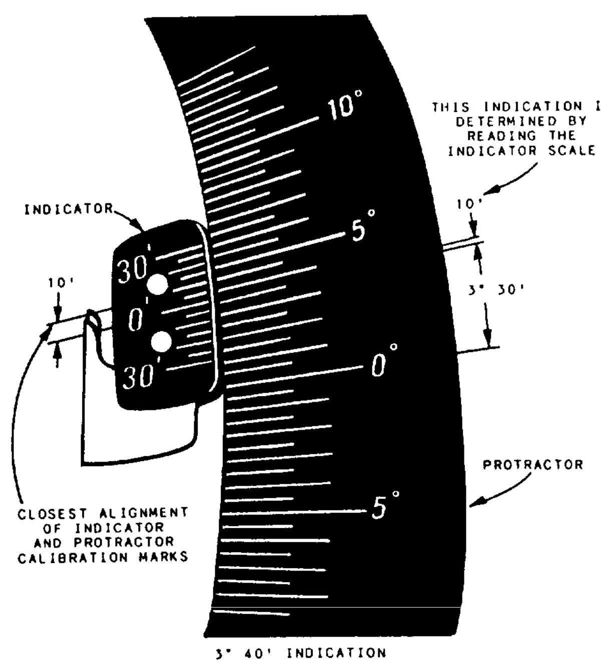


Figure 9-19.—Aileron throw protractor indications.

CABLE MAINTENANCE.—Cable control systems require more maintenance than rigid linkage systems; therefore, they must be inspected more thoroughly. Cables must be kept clean and inspected periodically for broken wires, corrosion, kinking, and excessive wear.

Broken wires are most apt to occur in lengths of cable that pass over pulleys or through fairleads. On certain periodic inspections, cables are checked for broken wires by passing a cloth along the length of the cable. Where the cloth snags the cable is an indication of one or more broken wires.

WARNING

Your bare hands should NEVER be used to check for broken wires. Using your bare hands to check for broken wires could result in personal injury.

Tests have proven that control cables may have broken wires and still be capable of carrying their designated load. However, any 7 x 19 cable that shows more than six broken wires in any 1-inch length, or any 7 x 7 cable that shows more than three broken wires in any 1-inch length, must be replaced. A maximum of three broken wires per inch is allowable in the length of cables passing over pulleys, drums, or through fairleads. Figure 9-20 shows how to determine if a cable is serviceable.

Corrosion, kinking, and excessive wear should be given particular attention during cable inspection. If a cable is found to be kinked or badly worn, it should be replaced, even though the number of broken wires is less than that specified for replacement. If the surface of the cable is corroded, relieve the tension on the cable and carefully untwist it to visually inspect the interior. Any corrosion on the interior strands of the cable constitutes failure, and the cable must be replaced. If no internal corrosion is detected, remove loose, external corrosion with a clean, dry rag or fiber brush and apply the specified preservative compound.

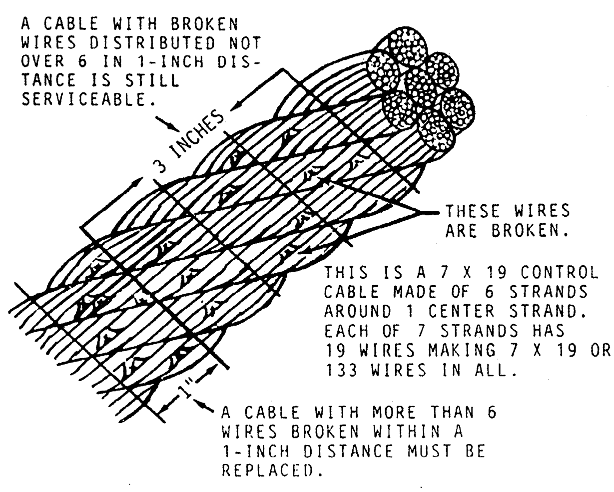


Figure 9-20.—Determining serviceable cable.

NOTE: Do not use metal wool or solvents to clean installed cable. Metal wool will embed tiny dissimilar metal particles and create further corrosion problems. The use of solvents will remove the internal cable lubricant and allow the cable strands to abrade and further corrode.

When a cable is found to be unserviceable and a spare cable is not available, an exact duplicate of the damaged cable may be prepared. This will involve cutting a length of cable to the proper length, attaching the necessary end fittings, and testing the assembly.

To determine the proper length to which the new cable will be cut, you should first determine the overall length of the finished cable assembly. This may be accomplished by measuring the old cable assembly or by reading the measurements provided in the MIM for the aircraft concerned.

Replacing cables in the aircraft, especially those routed through inaccessible spaces, can be difficult. One method is to secure a snaking line to the cable to be replaced, remove the pulleys from the brackets, and pull out the old cable while pulling the snaking line into the cable system run at the same time. Attach the new cable assembly to the snaking line, and pull the snaking line out to pull the new assembly into place. Replace the pulleys and attach the new cable in the system.

QUICK DISCONNECTS.—Quick disconnects are used in cable systems that may require frequent disconnecting. One type of quick disconnect is made with steel balls swaged to the ends of the cable, slipped into a slotted bar, and secured with spring-loaded sleeves on each end of the bar. Figure 9-21 shows the procedures for disconnecting and connecting this type of quick-disconnect fitting.

Rigid Control Systems

Rigid control systems transfer useful movement through a system of push-pull rods, bell cranks, walking beams, idler arms, and bungees. The simplest rigid control system may consist of push-pull rods and bell cranks only.

PUSH-PULL RODS.—Push-pull rods are rigid tubes equipped with eye fittings at each end or with a clevis fitting at one end and an eye fitting at the other

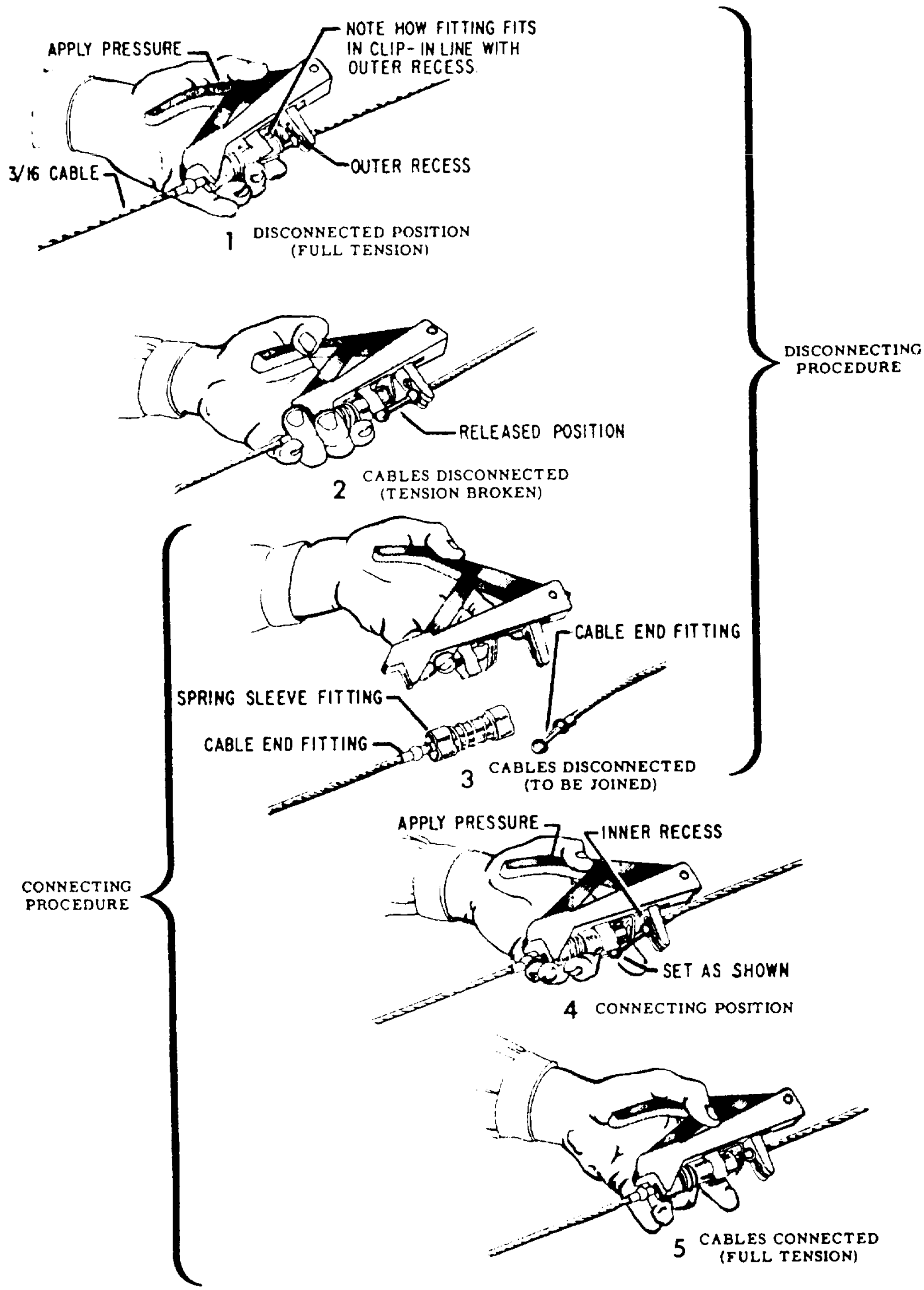


Figure 9-21.—Quick-disconnect procedures.

The eyes contain a pressed-in bearing. The rods are generally hollow and neck down to a smaller diameter at each end where the fittings are attached. One or both of the fittings are screwed into the necked portion of the rod, and are held in place by locknuts. When only one stem is adjustable, the stem of the other eye fitting is riveted into the neck at its end of the rod. A hole is drilled into the threaded neck of a push-pull rod for inspection to ensure that the stem has engaged a safe number of threads. The stem must be visible through the hole. Push-pull rods are generally made in short lengths to prevent bending under compression loads and vibration.

Push-pull rod linkage must be inspected closely for dents, cracks, and bent tubing. Damaged tubes may have to be replaced. End fittings are checked for damage, wear, and security of attachment. Worn or loose fittings must be replaced.

When you are replacing a damaged push-pull tube, the correct length of the new tube may be obtained by loosening the check nut and turning the end fitting in or out, as necessary. When the push-pull tube has been adjusted to its correct length, the check nut must be tightened against the shoulder of the end fitting. Normally, only one end of a push-pull rod is adjustable. The adjustable end has a hole (witness hole) drilled in the rod. The hole is located at the maximum distance the base of the end fitting is allowed to be extended. If the threads of the end fitting can be seen through this hole, the end fitting is within safe limits.

When you are attaching push-pull rods with ball bearing end fittings, the attaching bolt and nut must tightly clamp the inner race of the bearing to the bell crank, idler arm, or other supporting structure. Nuts should be tightened to the torque values listed in the aircraft MIM.

After installing a new push-pull rod in a flight control system, the control surface must be checked for correct travel. Procedures for accomplishing this are described later in this chapter. If the travel is incorrect, the length of the push-pull fod must be readjusted.

BELL CRANKS AND WALKING BEAMS.—Bell cranks and walking beams are levers used in rigid control systems to gain mechanical advantage. They are also used to change the direction of motion in the system when parts of the airframe structure do not permit a straight run. They are often used in push-pull tube systems to decrease the length of the individual tubes, and thus add rigidity to the system.

A bell crank has two arms that form an angle of less than 180 degrees, with a pivot point where the two arms meet. The walking beam is a straight beam with a pivot point in the center. Bell cranks and walking beams are mounted in the structure in much the same way as pulley assemblies. Brackets or the structure itself may be used as the point of attachment for the shaft or bolt on which the unit is mounted. Examples of a bell crank and a walking beam are shown in figure 9-22. The two are similar in construction and use. The names bell crank and walking beam are often used interchangeably.

IDLER ARMS.—Idler arms are levers with one end attached to the aircraft structure so it will pivot and the other end attached to push-pull tubes. Idler arms are used to support push-pull tubes and guide them through holes in structural members.

BUNGEE.—Bungees are tension devices used in some rigid systems that are subject to a degree of shock or overloading. They are similar to push-pull rods, and perform essentially the same function except that one of the fittings is spring-loaded in one or both directions. That is, a load may press so hard (compression) against the fittings that the bungee spring will yield and take up the load. This protects the rest of the rigid system against damage. The internal spring may also be mounted to resist tension rather than compression. An internal double-spring arrangement will result in a bungee that protects against both overtension and overcompression.

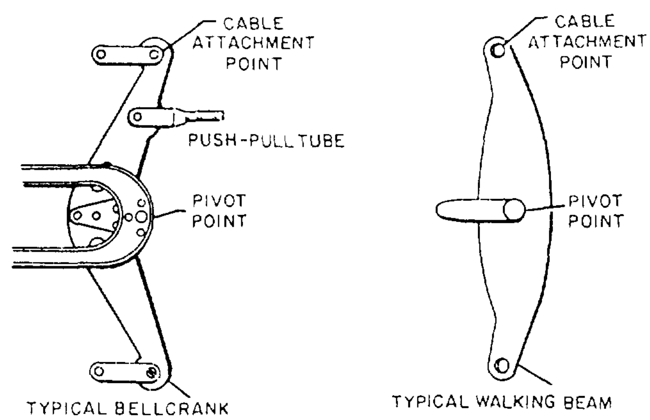


Figure 9-22.—Bell crank and walking beam.

CABLE AND RIGID CONTROL SYSTEMS TROUBLESHOOTING

When the cause and remedy for a reported malfunction in a control system are not immediately obvious to you, it maybe necessary to troubleshoot the system. Most aircraft MIMs provide troubleshooting charts that list some of the more common malfunctions in a system. Each discrepancy is accompanied by one or more probable causes, and a remedy is prescribed for each cause. The troubleshooting charts are organized in a definite sequence under each possible trouble, according to the probability of failure and ease of investigation. To obtain maximum value from these charts, they should be used systematically according to the aircraft manufacturer's recommendations. Examples of typical troubleshooting charts and instructions on their proper use was discussed in chapter 3 of this TRAMAN.

Since most aircraft use some form of electrical control or hydraulic boost in their flight control systems, maintenance of these systems must include the related electrical circuits and hydrauIic systems. Although an AE or AM is generally called upon to locate the correct electrical or hydraulic troubles respectively, you should be able to check circuits for loose connection, perform continuity checks, and perform minor troubleshooting of the hydraulic system.

Basically there are seven distinct steps to follow during troubleshooting. These steps were discussed in chapter 3 of this TRAMAN.

RIGGING AND ADJUSTING TOOLS

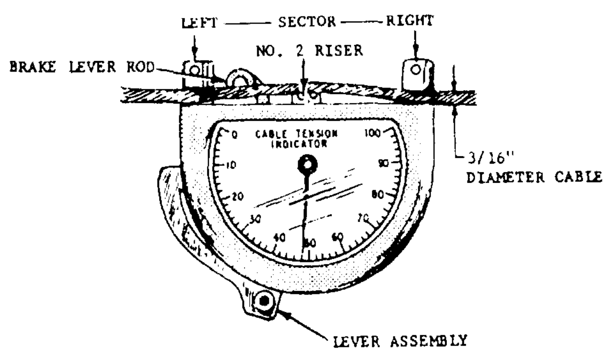
The purpose of rigging and adjusting a primary flight control system is to ensure neutral alignment of all connecting components and to regulate and limit the surface deflection in both directions. Each aircraft has a set of special tools for flight control maintenance that may include rigging fixtures, pins, blocks, throwboards and protractors. Other common equipment, such as micrometers, pressure gauges, push-pull gauges, feeler gauges, tensiometer and calipers may also be required. These arc usually maintained in the toolroom and checked out when needed.

Tensiometer

The tensiometer is an instrument used in checking cable tension. Tension is the amount of pulling force applied to the cable. The amount of tension applied in a cable linkage system is controlled by turnbuckles in the system.

A tensiometer is a precision cable tension measuring device, but it has limitations and can be awkward to use. It is inaccurate for cable tension under 30 pounds. When you take tension measurements, the instrument must not be pressed against any part of the aircraft, it can't be pushed or pulled against the cable, and the cable must not be pressed against fairleads or any part of the aircraft. Any one of these actions may lead to inaccurate measurements. A major advantage of cable linkage is its minimal space requirement and the ease in which it can be routed around, through, and behind aircraft structures and components. This can make access difficult and the tensiometer awkward or difficult to use. Adequate clearance for the tensiometer is necessary. All tensiometers must be certified by a calibration laboratory for accuracy at least once a month.

One type of tensiometer is shown in figure 9-23. This instrument works on the principle of measuring



No. 1				Riser	No. 2		No. 3	
DIA	1/16	3/32	1/8	Tension Lbs.	5/32	3/16	7/32	1/4
	14	15	21	30	14	20		
	18	22	27	40	18	25		
	23	27	33	50	22	29		
	27	32	39	60	26	33		
	31	37	44	70	29	37		
	35	41	49	80	32	41		
	39	45	54	90	35	45		
	43	49	59	$\boxed{100}$	38	$\boxed{48}$		
	46	53	63	110	41	52		
	49	57	67	120	44	55		
	53	61	71	130	47	58		
	56	65	75	140	49	61		
	59	68	79	150	51	64		
	62	71	82	160	54	67		
	65	74	86	170	56	70		
	68	77	90	180	58	72		
	70	80	93	190	60	74		
	72	83	96	200	62	76		
				220	66	81		
				240	70	85		
				260	74	89		
				280	78	93		
				300	80	97		

Typed figures are instrument scale reading corresponding to tension

Instrument No. 6659
Model 401-1C-2

Figure 9-23.—Cable tensiometer and chart.

the amount of force required to deflect a cable a certain distance at right angles to its axis. The cable to be tested is placed under the two blocks on the instrument, and the lever assembly on the side of the instrument is pulled down. Movement of this lever pushes up on the center block, called a "riser." The riser pushes the cable at right angles to the two clamping points. The force required to do this is indicated by a pointer on the dial. Different risers are used with different size cables. Each riser carries an identifying number, and is easily inserted in the instrument.

Each tensiometer is supplied with a calibration table to convert the dial readings into pounds. One of these calibration tables is shown in figure 9-23. For example, if the pointer on the dial indicates 48 with a No. 2 riser and a 3/16-inch diameter cable, the actual tension on the cable is 100 pounds. With this particular instrument, the No. 1 riser is used with 1/16-, 3/32-, and 1/8-inch diameter cables.

CAUTION

The calibration table applies to the particular instrument only, and cannot be used with any other. For this reason, the calibration table is secured inside the cover of the box in which the instrument is kept. The chart is serialized with the same serial number as the instrument. Using the calibration table from another instrument will result in inaccurate reading.

During the adjustment of turnbuckles, the calibration table must be used to obtain the desired tension in a cable. For example, to obtain a tension of 110 pounds in a 3/16-inch diameter cable, the No. 2 riser is inserted in the instrument and the number opposite 110 pounds is read from the calibration table. In this case, the number is 52. The turnbuckle is then adjusted until the pointer indicates 52 on the dial.

NOTE: Tensiometer readings should not be taken within 6 inches of any turnbuckle, end fitting, or quick disconnect.

In some cases, the position of the tensiometer on the cable may be such that the face of the dial cannot be seen by the operator. In such cases, after the lever has been set and the pointer moved on the dial, the

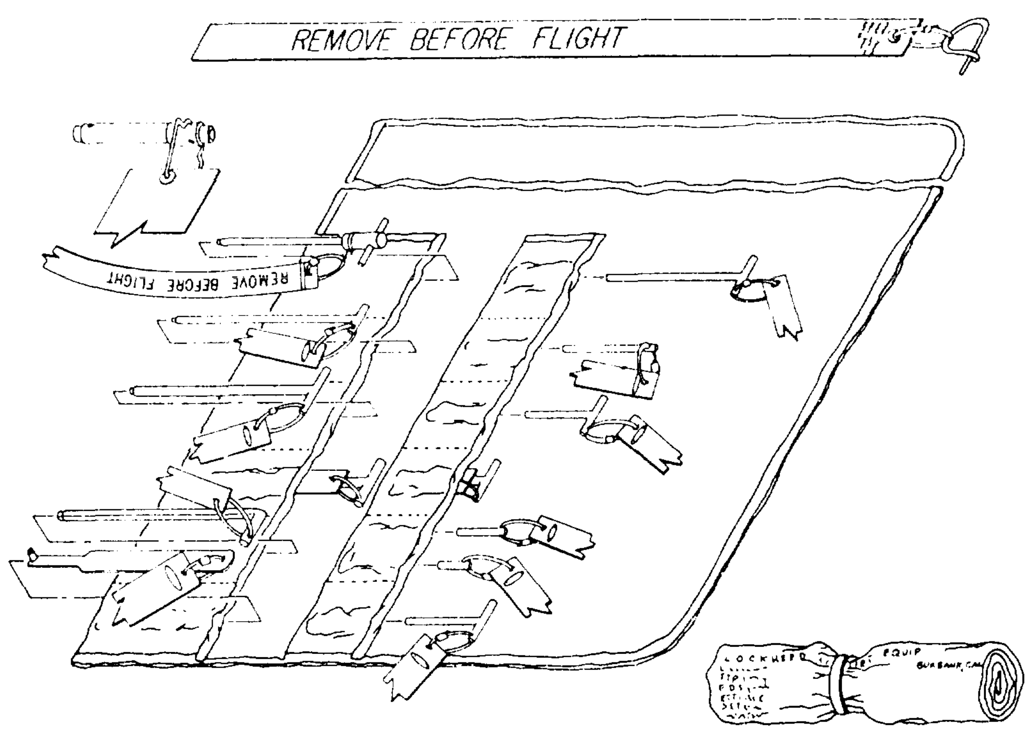


Figure 9-24.—Rigging pin set.

brake-lever rod on the top of the instrument is moved to the closed position. This locks the pointer in place. Then, the lever assembly is released and the instrument removed from the cable with the pointer locked in position. After the reading has been noted, the brake-lever rod is moved to the open position, and the pointer will return to zero.

The tensiometer, like any other measuring instrument, is a delicate piece of equipment and should be handled carefully. Tensiometers should never be stored in a toolbox.

Temperature changes must be considered in cable-type systems since this will affect cable tensions. When a temperature is encountered that is lower than that at which the aircraft was rigged, the cables become slack because the aircraft structure contracts more than the cables. When temperatures higher than that at which the aircraft was rigged are encountered, the aircraft structure expands more than the cables and tension is increased.

The cables in any cable linkage system are rigged according to a temperature chart that is contained in the applicable maintenance instructions manual. This chart will give the proper tensions for the various temperature changes above and below the temperature at which the system was rigged.

Rig Pins

Rig pins are used in rigging control systems. Figure 9-24 shows a rigging pin kit used on one of the Navy's aircraft. As you can see, rig pins may come in various sizes and shapes and may be designed for one or many installations. You should refer to the specific maintenance instructions manual for use and selection of rig pins.

Throwboards

Throwboards are special equipment used on specific aircraft for accurate measurement of control surface travel. See figure 9-25. Each throwboard has a protractor scale that indicates a range of travel in degrees. Zero degrees normally indicates the neutral position of the control surface. When the throwboard is mounted and the control column or stick is in neutral, the trailing edge of the control surface should be aligned to zero. As the control column or stick is moved to its extreme limits, you can read the corresponding degree indication on the throwboard. If the travel of the control surface is out of limits, you should adjust cables, push-pull rods, and control limit stops to obtain the correct control surface travel. When you are inspecting and rigging control surfaces, the specific maintenance instructions manual should be consulted.

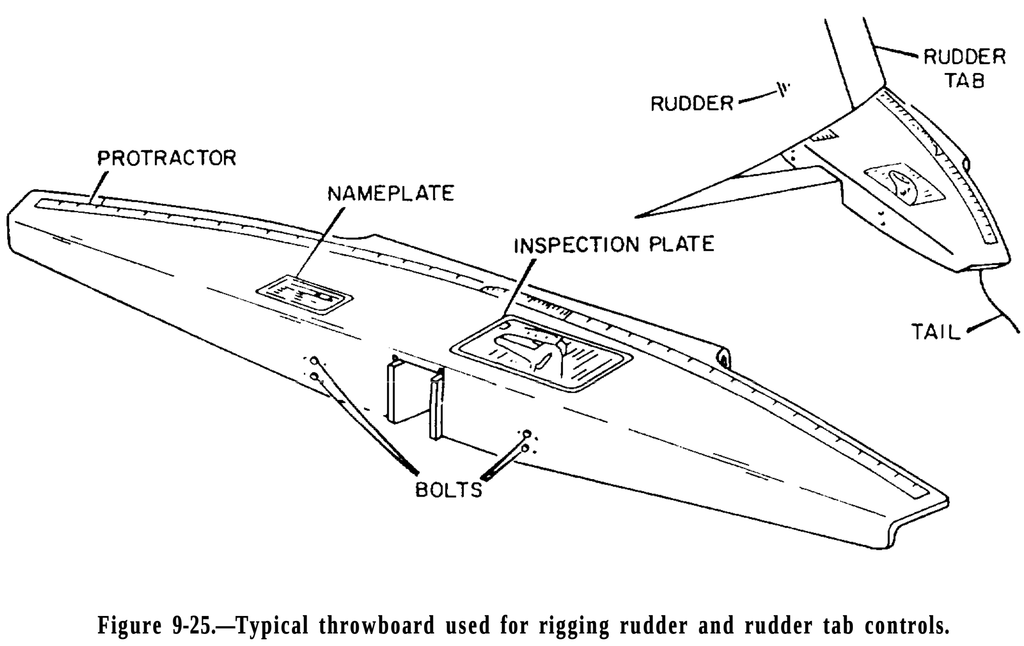
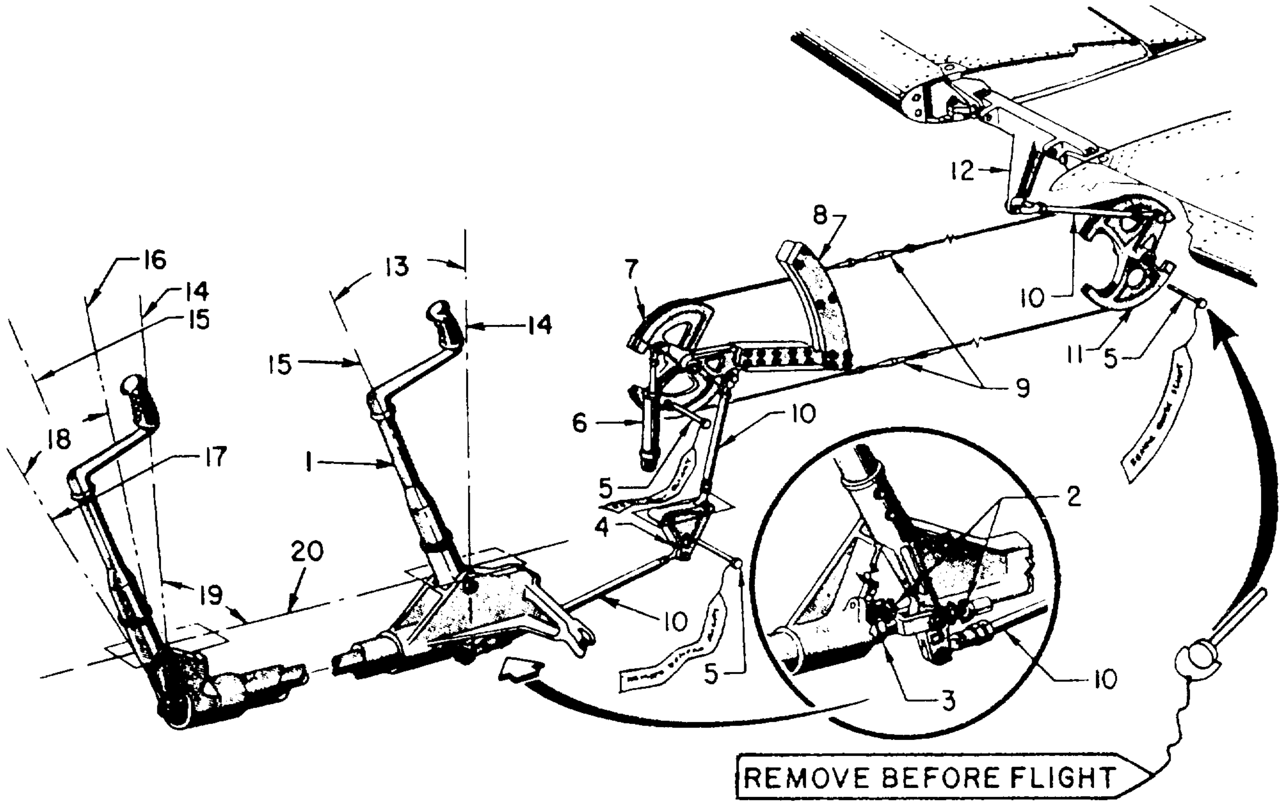


Figure 9-25.—Typical throwboard used for rigging rudder and rudder tab controls.



1. Aft control stick
2. Stop bolts
3. Push-pull tube adjustment
4. Bell crank
5. Rig pins (3 places)
6. Bungee
7. Forward sector
8. Bobweight
9. Turnbuckles
10. Push-pull tubes
11. Aft sector
12. Elevator fitting assembly
13. Rigging dimension
14. Vertical references line
15. Center line-stick neutral
16. Stick throw limit-UP elevator
17. Stick throw limit-DOWN elevator
18. Stick throw range-elevator control
19. Locating angle-vertical reference line
20. Longitudinal reference line (cockpit floor)

Figure 9-26.—Typical elevator flight control system.

CABLE AND RIGID CONTROL SYSTEMS RIGGING

In the elevator system shown in figure 9-26, rigging begins at the aft sector. The aircraft manufacturer has determined the position of the aft sector when it is in the neutral position. A rig pinhole has been furnished in the sector and a mating hole in the adjoining structure. See the three rig pins in figure 9-26. With the rig pin inserted in the aft sector and in the aircraft structure, the sector is held firmly in the neutral position. With the sector in this position, the push-pull tube connecting the sector with the elevator fitting assembly is adjusted to position the elevators to the neutral position. The neutral position is determined by using the elevator rigging fixture shown in figure 9-27. The curved section of the rigging fixture is graduated in degrees on either side of the neutral (zero degree) position that is about midway on the curved part of the fixture.

The rigging fixture is fastened securely to the aircraft at the indicated points of attachment. When properly mounted, the index marks (graduations) on the curved section align with the elevators and indicate the position, in degrees, of the elevators. If, with the aft sector rig pin in place, the elevators are not in neutral (for example, 5 degrees above the neutral mark), lengthening the push-pull rod end will push the elevator fitting assembly forward, and thereby lower the elevators. If the elevators are too low, then shortening the rod will bring them up as required.

The next step is the adjusting and tightening of the pair of cables in the system. This is accomplished by tightening the turnbuckles on each cable evenly until the required tension is obtained. During cable tightening, the rig pin is retained in the aft sector, leaving the forward sector free to turn. Therefore, when the necessary tension is recorded on one cable, that is also the tension on the other cable. To ensure that the cables were tightened evenly, check the forward sector rig pin hole to see if the rig pin can be inserted through the sector and into the structure. If this is not possible, then the cables must be adjusted by loosening one and tightening the other. This will maintain the correct tension on the cables, and, at the same time, rotate the forward sector to the neutral position. The cable section is properly rigged when it is possible to insert and remove the forward sector rig pin easily with the aft sector pin installed and the cables tightened to the prescribed tension.

The push-pull rod connecting the forward sector and the bell crank is adjusted to the correct length by installing a rig pin in the bell crank. Then, the rod adjustable eye is turned in or out until the rod can be installed between the sector and bell crank without binding. At this point three rig pins are in place, and should remain in place until the control sticks are rigged to neutral.

When you are positioning the control sticks to neutral, the rear stick must be adjusted first. Remember, we are working forward from the elevator surface. The push-pull rod connecting the bottom of the rear stick with the bell crank must be adjusted until the stick center line is the prescribed number of degrees forward of a vertical reference line. See the vertical reference line (14) and the center line (15) in figure 9-26. The vertical reference line is a position that the center line of the control stick would attain at a 90-degree angle (19) to the cockpit floor (20).

Adjust the length of the push-pull tube between the control sticks to position the front control stick to an angle identical to that of the aft control stick. Then, remove all three rig pins. This completes the rigging and adjusting of the control system to neutral. All that remains is to adjust the stops that limit the fore and aft

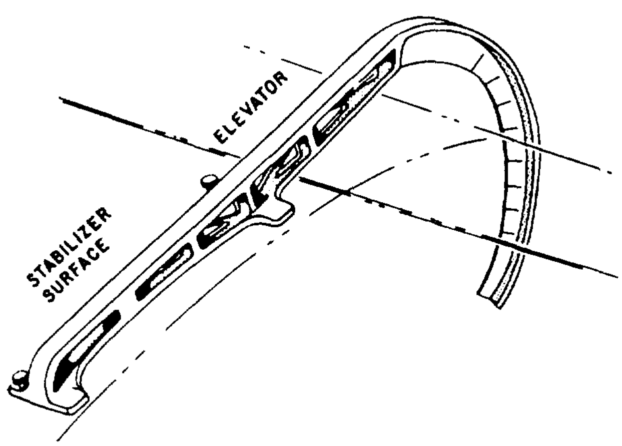


Figure 9-27.—Elevator rigging fixture.

travel of the control sticks, and rig and adjust the bungee that holds the system in the neutral position.

The stop bolts (2) (fig. 9-26) are located in front and behind the aft control stick. They are installed so that the stick hits the stop bolts at the extreme limits of its travel. The maximum travel of the elevators in each direction is determined by the manufacturer and is controlled by the stop bolts. With the rigging fixture still in place, move the control stick all the way forward, and adjust the stop until the elevator DOWN throw conforms to the MIM. Pull the stick all the way aft, and adjust the aft stop bolt to obtain the correct elevator UP throw. The stop bolts are safety wired in place after this adjustment.

The last item to be adjusted in this control system is the centering bungee. Connect the bungee and adjust its rod end so that with the stick against the stop bolt in the full down elevator position, the bungee is a minimum of 1/32 of an inch from bottoming. After this adjustment, the elevators should be held in neutral (plus or minus the prescribed number of degrees) by bungee action. If the elevators are too high, shorten the bungee rod end. If they are too low, lengthen the bungee. With the bungee properly adjusted, tighten the bungee rod end locknut and safety wire it.

CABLE FABRICATION

Control cables are fabricated mostly of extra flexible, preformed, corrosion-resistant steel. Control cables vary from 1/16 to 3/8 inch in diameter. Cables of 1/8 inch and larger are composed of 7 strands of 19 wires each. Cables 1/16 and 3/32 inch in diameter are composed of 7 strands of 7 wires each.

Cable-Cutting Equipment

Cutting cables may be accomplished by any convenient method except an oxyacetylene cutting torch. The method of cutting usually depends upon the tools and machines available. If a cable tends to unravel, the ends may be sweat soldered or wrapped with a strip of tape prior to cutting.

Small diameter cable may be cut satisfactorily with a pair of heavy-duty diagonal cutters, side cutters, or a pair of wire nippers. Best results are obtained if the cutting jaws are held perpendicular to the cable during the cutting operation. Cables up to 3/32 of an inch in diameter may be cut in one operation by this method. Larger cables may require two or more cuts. When you cut large diameter cables, use the end of the cutting blade, and cut only a few strands at a time.

The most satisfactory method of cutting cables is with a cable-cutting machine that has special jaws to accommodate various sizes of cable. See figure 9-28. To use this machine, position the cable in the proper diameter groove and hold the cable firmly within 2 inches of the cutting blades. Hold the cable at right angles to the cutting blades and pull the operating handle down sharply.

A cold chisel and a soft metal block may also be used for cutting cables. This method should be used only as a last resort because of the way the cable ends will be frayed.

Terminal Swaging

After the cable is cut, the next step in making up an aircraft cable is attachment of the terminals. Most terminal fittings are SWAGED onto the ends of control system cables. Swaging is essentially a squeezing process in which the cable is inserted into the barrel of the terminal. Then pressure is applied by dies in a swaging machine to compress the barrel of the terminal tightly around the cable. The metal of the inside walls of the barrel is molded and cold flowed by force into the crevices of the cable. Figure 9-29 shows two types of hand-swaging tools. The one in the upper part of the illustration is mechanically operated, while the lower one is pneumatically operated.

When you prepare to swage a terminal, cut the cable to the required length. Be sure to allow for the elongation (increase in length due to stretching) of the fitting that will occur during the swaging process. The amount of elongation will vary with the type and size of fitting used. Therefore, the elongation must be taken into account whenever you make up any cable. The *Structural Hardware Manual,* NAVAIR 01-1A-8, provides elongation data for all types and sizes of fittings.

Make sure that the cable end is cut square and clean and that all strands remain in a compact group, as shown in figure 9-30. Place a drop or two of light lubricating oil on the cable end. Then, insert the end into the terminal to a depth of about 1 inch. Bend the cable toward the terminal, straighten it back to the normal position, and then push the cable all the way into the terminal barrel. This bending process puts a kink in the cable end to hold the terminal in place until the swaging operation is completed. It also tends to separate and spread the strands inside the terminal barrel and reduces the strain caused by swaging.

Both of the hand-swaging tools shown in figure 9-29 are widely used by naval aircraft maintenance activities. The procedure for using both types is described in the following paragraphs.

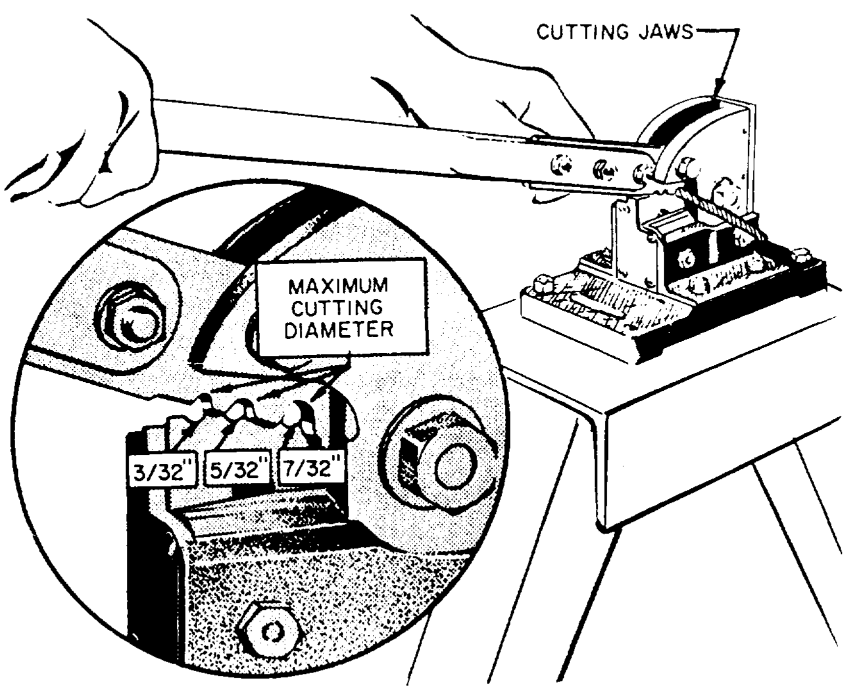


Figure 9-28.—Cable-cutting machine.

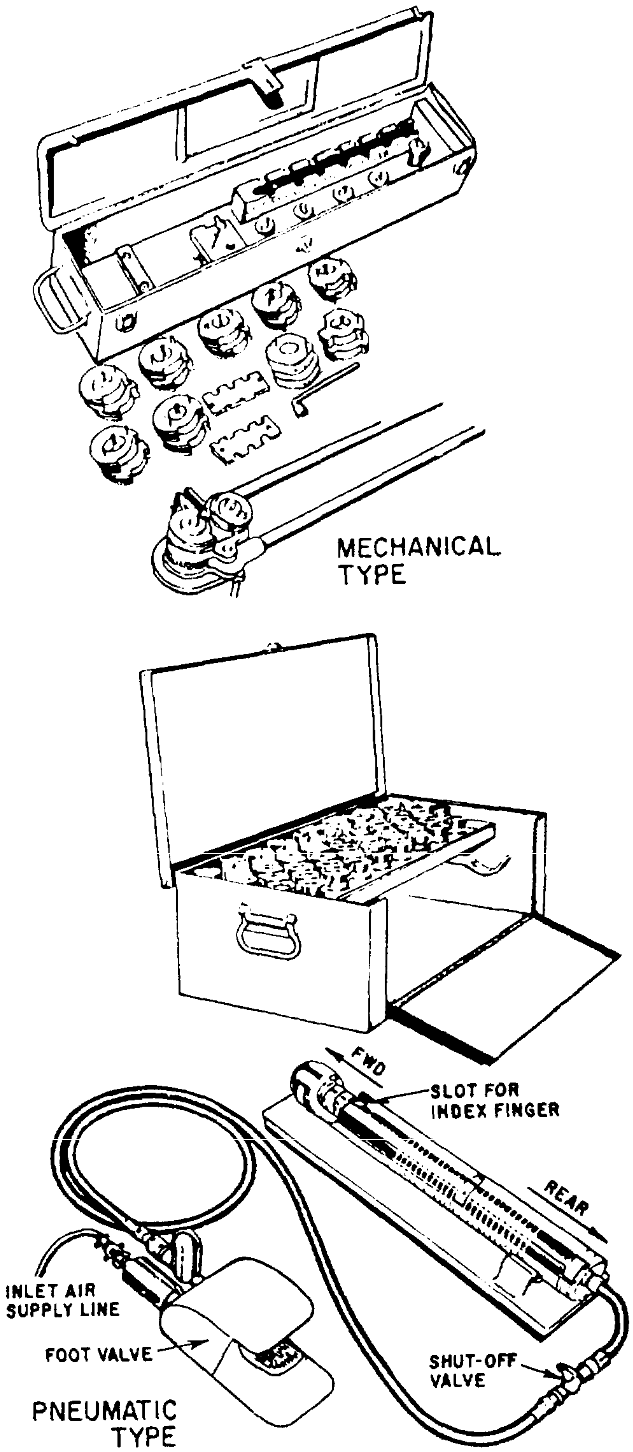


Figure 9-29.—Hand-swaging tools-mechanical and pneumatic.

When operating the mechanical swaging tool, you should place the proper size pair of dies on the swaging tool. The terminal is then located in the jaws of the tool, as shown in figure 9-31, and the swaging operation is performed. As the dies rotate, they pull

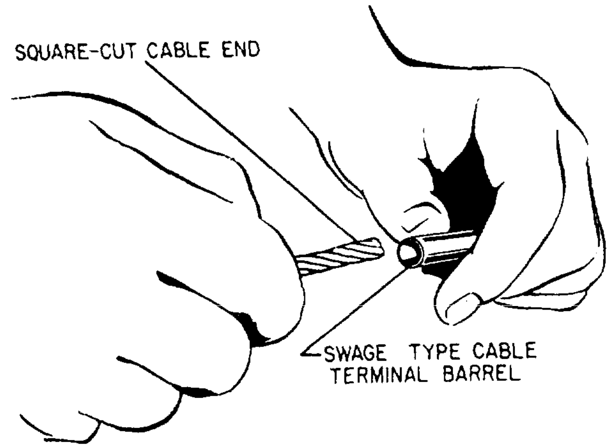


Figure 9-30.—Inserting cable in swage terminal.

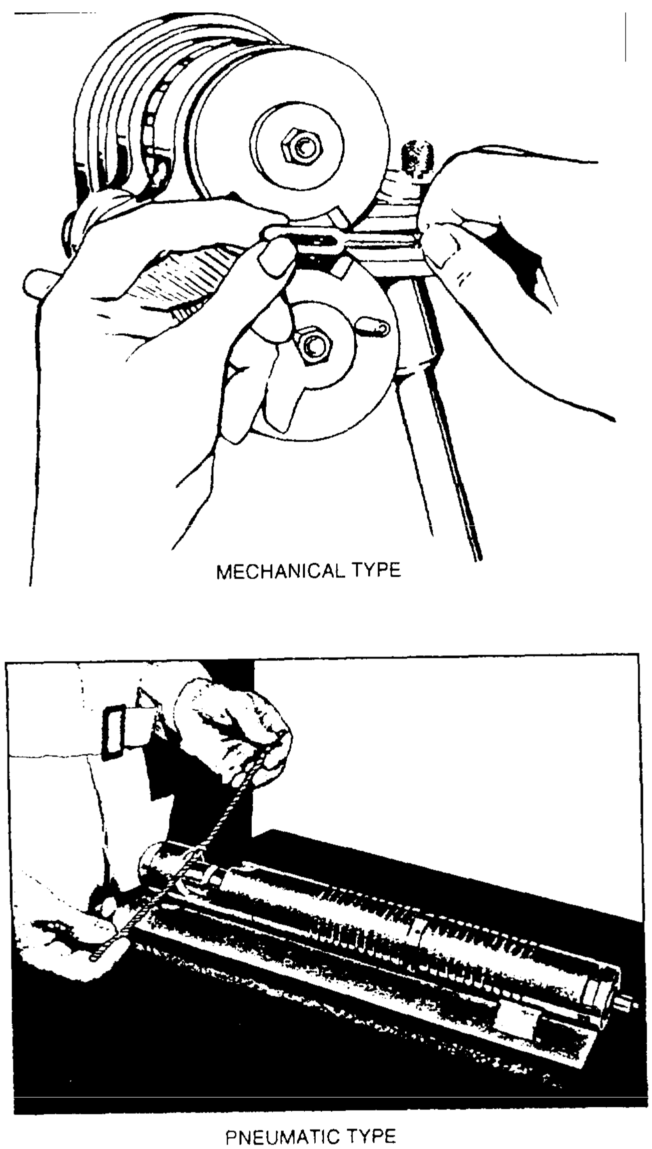


Figure 9-31.—Locating the terminal in the swaging tool.

the terminal from right to left. The dies compress the terminal barrel onto the cable, and swaging occurs. Rotation of the dies is accomplished by opening and closing the handles.

After completion of swaging and removal of the fitting from the swaging tool, measure the outside diameter of the shank with a micrometer or with the gauge furnished with the swaging outfit to determine whether or not the terminal has been swaged sufficiently. This may be determined by checking the measurement with the applicable cable terminal table in NAVAIR 01-1A-8.

The pneumatic swaging tool shown in figure 9-29 is a lightweight portable unit designed to precision swage the metal of a terminal into the interstices (crevices) of the cable strands. The swager may be mounted on a baseplate and used on a bench, or it can be taken to the job. When the swaging tool is taken to the location of the job, it may be held in your hand or cradled in your arm.

The pneumatic swaging kit has several different sizes and types of dies used for swaging ball-and-sleeve terminals and for cutting and trimming cable. Like the mechanical swaging tool, the dies come in matched sets and must be used together. The dies are installed by inserting either die through the yoke opening into the die cavity. The keyway should be down and the shank facing the rear of the swager. Slide the first die back in order to clear the opening for the insertion of the mating die. The second die is inserted with the shank facing forward.

The following step-by-step procedures are recommended for setting up the pneumatic swaging tool :

1. Connect the air supply to the foot valve. For efficient operation, use an inlet air line with at least 3/8-inch inside diameter and a minimum of 90 pounds of line pressure.

2. Connect the swager air line to the foot valve.

3. Clean the dies, remove any steel particles that may have adhered to the die cavity, and apply a light film of oil to the entire die.

4. Insert the dies in the swaging tool as previously described.

WARNING

Do not insert or remove dies until the air supply that is connected to the swager is shut off. Failure to secure the air supply connected to the swager could result in personal injury to the operator.

With the pneumatic tool set up for use, perform the following steps while swaging terminals to cables:

1. Position the terminal on the cable, using the old cable as a pattern, or follow the instructions given in the applicable technical directives. When you are using a ball terminal, a minimum of 1 1/2 inches of cable must extend beyond the ball to allow room for holding and turning the terminal during swaging. The excess is trimmed, if necessary, after the swaging operation. When you use MS 20667 terminals, 1/4 inch of cable must extend through the terminal. On all other terminals, the cable is bottomed (inserted all of the way into the terminal).

2. Each terminal is cleaned with a suitable solvent, and then coated with a light oil.

3. With the terminals positioned in the cavity of the forward die, slide the rear die to its forward position using the slot provided in the yoke for the index finger.

NOTE: To prevent damage to terminal or cable during the swaging cycle, maintain light pressure on the cable towards the front of the swager. This holds the terminal and cable firmly in the forward die cavity.

4. Depress the foot valve firmly and rotate the cable back and forth in 180-degree arcs or complete revolutions. The length of time the foot valve is held depends upon the type and size of fitting being swaged. The proper time can be found by referring to the chart supplied with the pneumatic swaging tool. If the terminal will not rotate, stop swaging immediately; rotate the terminal 90 degrees, and start swaging again.

5. Release the foot pedal to stop swaging, and remove the terminal from the swaging tool for inspection. If the diameter is oversize or the terminal surface is too rough, repeat the operation.

If swaged terminals are to be used on both ends of the cable, recheck the overall length of the cable and trim it, if necessary, prior to installing the second terminal. Make certain that all additional fittings and accessories, such as cable stops and fairleads, are slipped onto the cable in the proper sequence. The other terminal may then be swaged, using the same procedures as used for the first one.

Proof-Testing Cables

All newly fabricated cables should be tested for proper strength before they are installed in aircraft The test consists of applying a specified tension load on the cable for a specified number of minutes. The proof loads for testing various size cables are given in tables contained in NAVAIR 01-1A-8. Proof loading will result in a certain amount of permanent stretch being imparted to the cable. This stretch must be taken into account when You fabricate cable assemblies. Cables that are made up slightly long may be entirely too long after proof loading.

SECONDARY FLIGHT CONTROL SYSTEMS

Learning Objective: *Recognize the varied functions of secondary flight control systems and the maintenance associated with each system.*

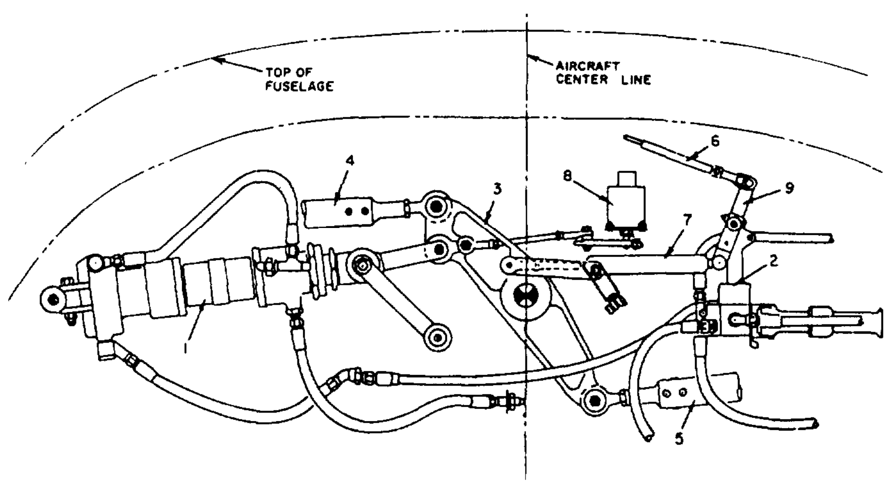
Secondary flight controls, such as wing flaps and speed brakes, are usually hydraulically operated and either mechanically or electrically controlled. The design of these flight controls slows the aircraft in flight and provides additional lift and stability. These design features greatly increase the versatility and performance of the aircraft.

CONVENTIONAL WING FLAP SYSTEM

A flap is a hinged or pivoted section that forms the rear portion of an airfoil used to vary the effective chamber. Wing flaps in their most commonly used form are hinged sections of the trailing edges of a wing. Flaps extend from the fuselage to the inboard side of the aileron. Wing flaps are connected to the main wing by various kinds of hinges and slides.

The flap system discussed in this section is a representative system. The number of flaps will vary according to the size of the aircraft. The components may have different names, depending on the manufacturer, but the operational theory remains the same. This system consists of a series of six flaps, three on the trailing edge of each wing. They raise and lower in the conventional manner by a hydraulically actuated linkage of bell cranks, pushrods, and idlers. The flap control lever in the cockpit controls the system mechanically. The lever connects by conventional and teleflex cables to the hydraulic actuating mechanism. An emergency system is provided for lowering the flaps by operating a hand pump if the primary system malfunctions. The flap system has a position indicator and several safety devices to prevent lowering of the flaps while the wings are folded, or folding of the wings while the flaps are lowered.

The movement of the flap selector lever in the cockpit sets the flaps in motion. Movement of the selector lever operates a cable quadrant to which a set of conventional control cables attach. These cables connect to another sector just forward of the main wing beam. A teleflex cable, also attached to this aft sector, and a spring-loaded pushrod on the main flap actuating bell crank connect to the two ends of a short floating arm installed on the hydraulic selector valve lever. Figure 9-32 is a drawing of the cylinder, linkage, and selector valve installation. Reference to the index numbers on this drawing is made in the following description of the operation of the flap control system.



1. Wing flap cylinder
2. Wing flaps selector valve
3. Flap actuating bell crank
4. Left flap control pushrod
5. Right flap control pushrod
6. Flap control push-pull cable assembly
7. Follow-up pushrod
8. Flap position transmitter
9. Selector valve floating arm assembly

Figure 9-32.—Flap cylinder, linkage, and selector valve installation.

When the flap handle in the cockpit moves down, the upper end of the floating arm (9) pulls to the left, pivoting at its lower end and moving the selector valve lever to the left. This action directs pressure from the hydraulic system to the flap actuating cylinder (1). The cylinder piston rod extends and lowers the flaps by rotating the flap drive bell crank (3) in a clockwise direction. As the bell crank moves, the lower end of the floating arm moves to the right by the spring-loaded pushrod (7). This action pivots the arm at its upper connection to the sector pushrod and returns the selector valve to neutral, stopping the action of the system.

Moving the flap handle upward reverses the foregoing procedure by pushing the selector valve lever to the right, directing hydraulic pressure to the retract side of the cylinder piston and raising the flaps. The follow-up rod then moves the lower end of the floating arm to the left and returns the selector valve to neutral, The valve will not return completely to neutral, maintaining pressure in the flap cylinder and ensuring positive locking of the flaps in the up position.

The spring mechanism in the follow-up rod normally does not function The spring mechanism is provided only as a safety feature, permitting actuation of the flap drive crank by emergency hydraulic power if the selector valve becomes jammed.

The flap hydraulic system consists primarily of the selector valve and the actuating cylinder. See figure 9-33. The selector valve is a four-way, poppet-type valve. The poppets operate in pairs to direct pressure to one side of the cylinder while opening the other side to reservoir return.

The cylinder is double acting and internally locked in the retracted (flaps up) position. The cylinder also has an integral shuttle valve (built into the mounting end cap). This provides for the separation between the normal and emergency hydraulic pressure lines. An adjustable terminal on the piston rod provides for length variation.

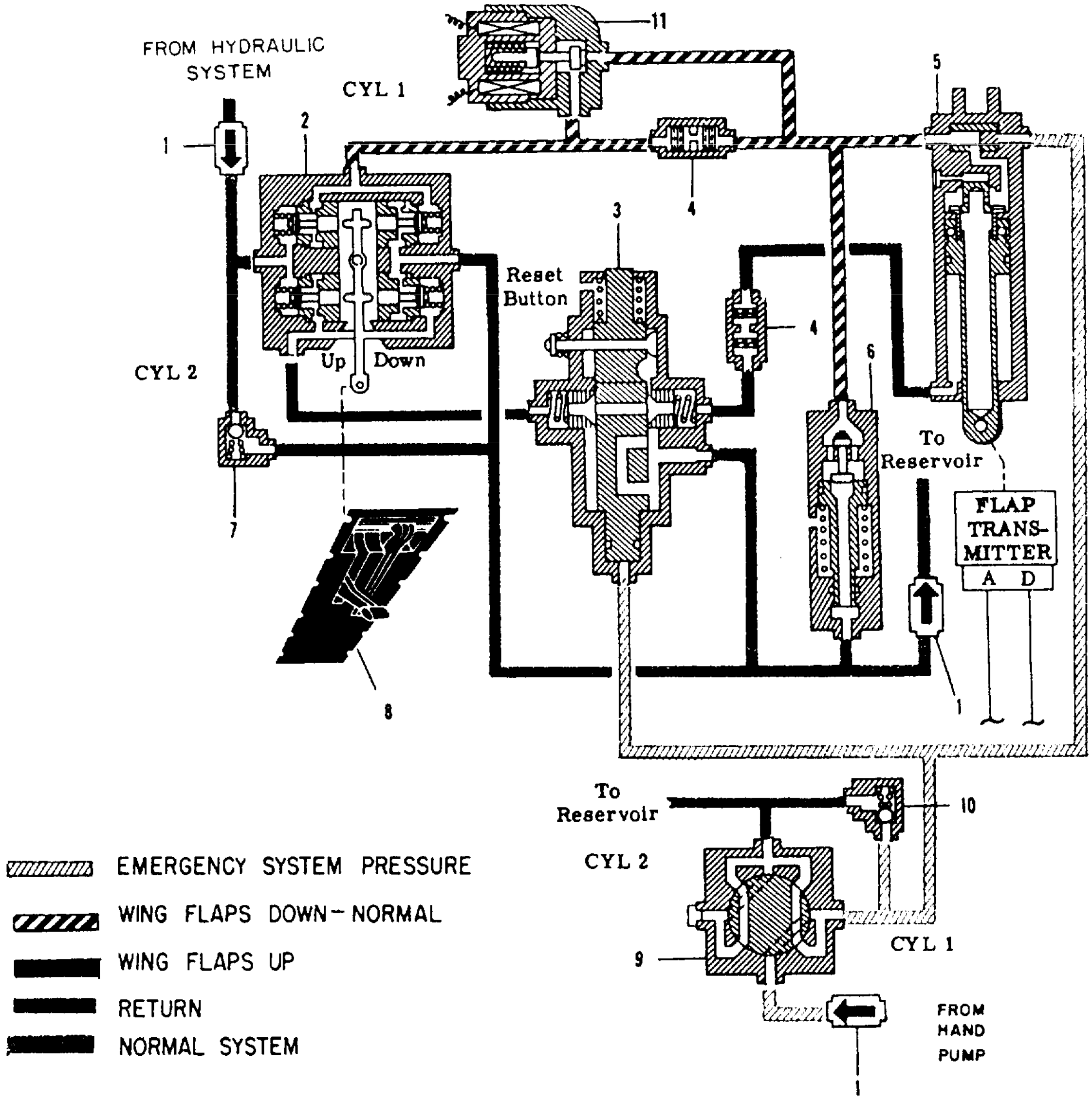
When the cylinder extends, the internal lock is hydraulically released, allowing the piston to move. When the flaps raise, the hydraulic pressure on the lock is relieved, and a compression spring engages the lock mechanism with the piston when the cylinder becomes fully retracted.

A relief valve installed in the normal flap down line provides a blowup feature that prevents overloading of the flaps and flap linkage. This valve is adjustable to a narrow range between full flow and reseat, providing a controlled blowup feature. As the flaps blow up, the flap air load decreases, gradually reseating the relief valve and preventing further flap retraction.

In the landing configuration, the flaps are partially or fully down. Safety microswitches prevent folding of the wings until the flaps are in the full up position. To reduce the recovery interval aboard ship, the aircraft wings must be folded and the aircraft taxied forward as quickly as possible. A wing flap retraction shutoff valve installed in the flap down line expedites flap retraction. This normally closed, solenoid-operated, hydraulic shutoff valve energizes only when the weight of the aircraft is on the wheels. When energized, the valve permits return fluid to bypass the restrictor in the down pressure line, permitting fast retraction of the flaps and quicker wing-fold operation.

A relief valve is located in the pressure line ahead of the flap normal system selector valve. The valve relieves pressure from thermal expansion, which may build up on the inlet side of the selector valve.

An emergency system for flap down operation includes a selector valve and an emergency dump valve. The emergency flap down selector valve is usually in the NORMAL position. In this position, the cylinder emergency line to return is vented. When you move the emergency selector valve handle to the FLAPS DOWN position, you can lower the flaps by operating the hand pump. This action directs hand pump pressure through the integral shuttle valve to the actuating cylinder. At the same time, the emergency dump valve is actuated. The emergency dump valve opens the up side of the cylinder directly to return and closes off its normal return line through the selector valve.



1. Check valve
2. Wing flap selector valve
3. Wing flap emergency dump valve
4. Restrictor
5. Wing flap cylinder
6. Wing flap blowup relief valve
7. Wing flap thermal relief valve
8. Wing flap control
9. Wing flap emergency selector valve
10. Relief valve
11. Wing flap snap shutoff valve

Figure 9-33.—Wing flap system.

Once actuated, the dump valve must be reset manual] y to restore the system to normal operation. The emergency selector valve handle must first be returned to the NORMAL position, relieving the pressure in the emergency line. The dump valve is then reset by pushing the button on the dump valve. The button is marked PUSH TO RESET. With pressure in the normal system, the normal selector handle must be placed in the down position to reset the integral shuttle valve. The flaps will then raise using normal control, provided the flap up portion of the system is operative. There are no provisions for emergency retraction of the flaps.

LEADING/TRAILING EDGE WING FLAP SYSTEMS

Several types of naval aircraft are equipped with flap systems that feature both leading edge and trailing edge flap panels. On some aircraft these leading edge panels are referred to as slats.

Figure 9-34 shows a leading edge and trailing edge flap arrangement. The figure shows flap operation with aileron drooping and boundary layer control. These features create even greater lift and stability than with flaps alone.

This flap system consists of three leading edge and one trailing edge flap panels for each wing, with each panel having its own actuator. A three-position flap control switch in the cockpit is labeled "UP, 1/2, and DN."

The leading edge flaps operate by a manifold-mounted selector valve and dual actuating cylinders. Trailing edge flaps use this same selector valve plus a wing-mounted selector valve and dual tandem actuating cylinders.

When the flap control switch is placed in the 1/2 position, the manifold-mounted selector valve directs utility system pressure through the shuttle valves. Pressure is sent into the down lines of the leading edge flap actuators. The leading edge flaps are lowered to the full down position. The inboard leading edge flap deflection is 30 +0, –2 degrees. The center flap deflection is 60 +0, –2 degrees. The outboard flap deflection is 55 1/2 degrees ±1/2 degree.

At the same time, hydraulic fluid flows through the fuselage-mounted flow divider and into the extend side of the dual tandem trailing edge flap actuating cylinder. This action moves the trailing edge flaps to the 1/2 position with a deflection of 30, ±2 degrees. The cockpit flap position indicator indicates barber poles while the flaps are in transit and flap position at the completion of selected movement. The limit switches are connected into the control circuit in series to provide an indication of flap position and to continuously energize the electrical circuits to maintain hydraulic pressure when the flaps are down.

Moving the flap control switch to the full down position actuates the wing-mounted selector valve, porting pressure through a second flow divider. Pressure is sent into the down side of the retracted half of the trailing edge flap cylinder, moving the flaps from the 1/2 to the full down position. Full down position is 60 +1, –2 degrees. Both flap position indicators will indicate DN when the cycle is completed.

Placing the flap control switch to the UP position allows hydraulic pressure to be directed to the retract side of all flap actuators. Position indicators indicate UP. The electrical control circuits and solenoids of both selector valves are de-energized.

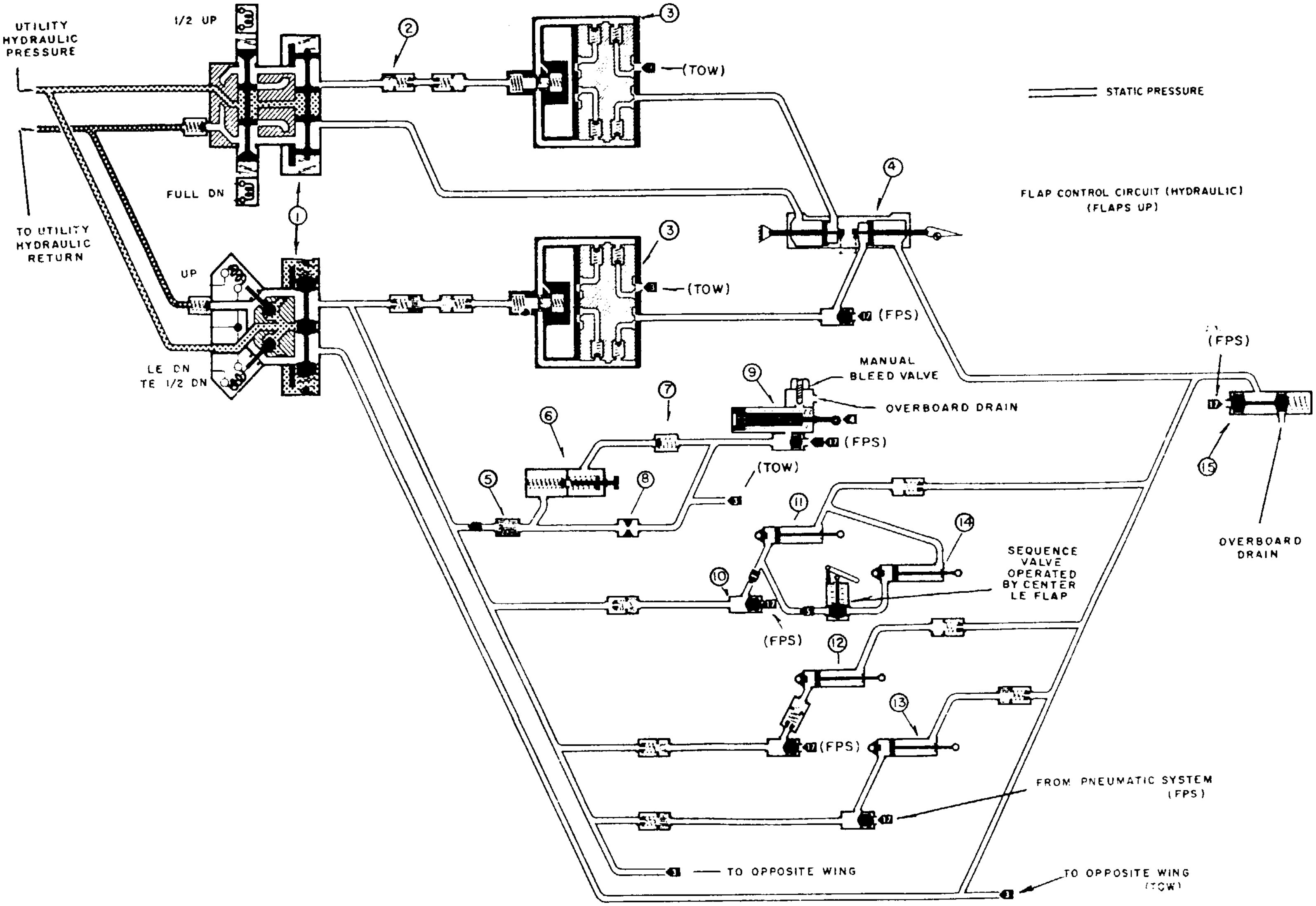
The leading edge flaps are locked in the UP position by the overcenter locking mechanism. The trailing edge flaps are locked up by internal locks within the trailing edge actuating cylinders.

HYDRAULIC DROOP AILERON SYSTEM

When the flap switch is placed in 1/2 or DN position, with PC 1, PC 2, and utility hydraulic power applied, the ailerons will extend 16 1/2 degrees down. The control stick will remain centered. The droop aileron actuating cylinder (fig. 9-34), one in each wing, extends by flap down utility hydraulic pressure. The droop aileron is retracted by springs in the cylinder when extend pressure is removed. The droop cylinder connects between the aircraft structure and an idler bell crank in the aileron power package linkage. With flaps up, the droop cylinder acts as a solid link. When the flap control switch is placed in the 1/2 or DN position, the droop aileron extend relay energizes. This relay completes the extend electrical circuit to the droop aileron actuators. As the actuators extend, the aileron power cylinder input levers reposition, and both ailerons droop as before. The actuators are de-energized by the integral extend limit switch. The ailerons are free to operate normally. When the flap control switch is placed to UP, the droop aileron extend relay is de-energized. The droop actuator reposition the aileron power cylinder input levers. Both ailerons move back to their normal position. The droop actuators are de-energized at the completion of the retract cycle by the integral limit switch.

EMERGENCY FLAP SYSTEM

If electrical and hydraulic power fails, the flaps can be lowered by the emergency system. An emergency flap extension bottle with a 300-cubic-inch capacity and charged to 3,000 psi provides a power source. Emergency extension is controlled by the emergency flap control handle, which is mechanically linked to the emergency flap air selector valve. Pulling the handle aft, the piston inside the air selector valve shifts, allowing high-pressure air to flow through a separate set of lines to shuttle valves in the flap system. The shuttle valves reposition, and air pressure extends the flap actuators. Air pressure also repositions the flap system dump valve, dumping return side hydraulic



1. Solenoid selector valves
2. One-way restrictor valve
3. Hydraulic flow divider
4. Trailing edge flap actuator (2)
5. Filter
6. Manual hydraulic bypass valve
7. Check valve
8. Two-way restrictor valve
9. Aileron droop actuating cylinder (2)
10. Shuttle valve
11. Inboard L.E. flap actuator
12. Center L.E. flap actuator
13. Outboard L.E. flap actuator
14. Boundary lay control valve actuator
15. Dump valve

Figure 9-34.—Flap control circuit.

fluid overboard. The leading edge flaps extend to the full down position and trailing edge flaps to the 1/2 down position. The aileron drooping feature does not operate when the flaps are lowered by the emergency flap system.

SEMI-INDEPENDENT FLAP AND SLAT SYSTEM

This system consists of semi-independent flap and slat systems, which raise and lower using hydraulic motors drive units, torque tubes, and screw jack-type actuators.

Flap System

The flaps divide into two panels per wing at the wing-fold joint. Each panel is supported by two sets of tracks and rollers that are driven by two ball screw actuators. Pressure from the combined hydraulic system powers the flap drive motor and gearbox assembly, shown in figure 9-35.

If the combined hydraulic system fails, a hydraulic brake locks the hydraulic motor, and an emergency electric motor provides continued operation. Emergency flap extension and retraction is controlled by placing the EMERG FLAP switch on the throttle quadrant at either UP or DN. Cam-operated switches within the flap drive gearbox provide input signals to show the flap position on the cockpit-integrated position indicator.

Operation of the flap control handle energizes the solenoid-operated flap selector valve, directing hydraulic pressure to the extend or retract lines of the flap drive motor. The wings must be spread and locked to provide a complete electrical circuit through the wing unlock relay to the selector valve.

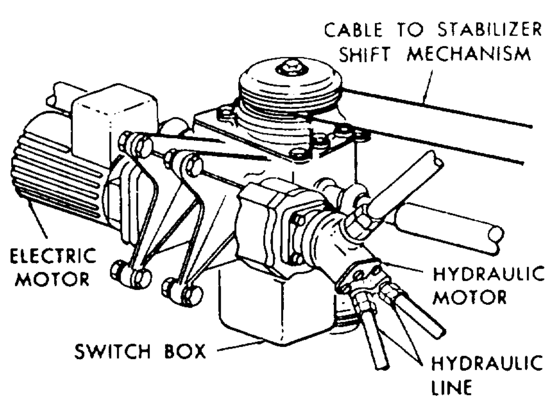


Figure 9-35.—Flap drive gearbox.

Placing the flap control handle to the TAKEOFF position completes the electrical circuit through the 30-degree switch and cam-operated flap drive gearbox limit switch to the selector valve. Pressure ports to the down side of the high-speed hydraulic motor, which drives the gearbox. The flap drive gearbox, through a series of torque tubes and offset gearboxes, drives all eight flap actuators.

The flap actuators, shown in figure 9-34, drive the carriage and attaching flaps out and down to the 30-degree position. The limit switch in the flap drive gearbox opens, de-energizing the selector valve circuit, allowing the valve shuttle to return to neutral, blocking flow to the motor, and preventing further flap extension.

Placing the flap control handle to LAND mechanically closes the 40-degree down flap handle switch. The electrical circuit to the selector valve completes, this time through the now closed 40-degree down limit switch in the flap drive gearbox. The flaps will extend to 40 degrees, and the electrical circuit will be broken by the action of the limit switch.

Moving the flap control handle to the TAKEOFF or UP position will energize the opposite solenoid of the flap selector valve and port pressure to the retract side of the flap hydraulic motor. If the TAKEOFF position is selected, a limit switch will again halt flap movement at the 30-degree position. If UP is selected, retraction will be halted when the flaps reach the full up position. Stopping the flaps is a function of the flaps up limit switch. At the same time, linkage from the up limit switch actuates a second switch to complete the electrical circuit to the flap hydraulic motor brake valve. The energized valve blocks combined hydraulic system pressure that is holding the hydraulic brake in the unlocked position. The brake locks the hydraulic motor, which, in turn, locks the flaps in the up position.

If combined hydraulic system pressure fails and the emergency flap switch is used, the flap action is powered by the electric motor. See figure 9-35. The flap hydraulic brake valve is energized, and the pressure holding the spring-loaded hydraulic motor brake unlocked will port to return. The brake is then free to lock the motor and input shaft.

The electric motor now drives the flap gearbox and associated linkage, bypassing the locked hydraulic motor. This action occurs until the flaps reach a 40-degree trailing edge down position. Limit

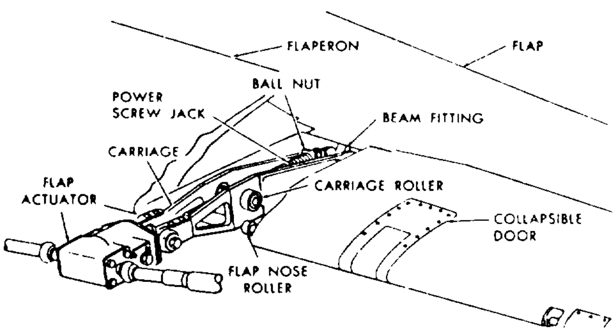


Figure 9-36.—Flap actuator.

switches shut the electric motor off when the flaps reach the 40-degree down and full up positions.

FLAP ACTUATOR.—The flap actuator shifts rotary motion of the input shaft to linear flap motion, using bevel gears and the ball screw jack mechanism. See figure 9-36. A load-sensing device in each flap actuator operates a clutch assembly to stall out the flap system if it is overloaded. An impact plate at the end of the ball screw (screw jack shaft) and mechanical stops on the actuator body protect the actuator against possible overtravel during flap extension and retraction.

OFFSET GEARBOXES.—The eight offset gearboxes in the flap system transmit power produced by the flap drive gearbox around wing structure obstacles and compensate for wing angularity. They also reduce the flap drive gearbox speed of 1,080 rpm to about 550 rpm at the outboard actuators.

FLAP WING-FOLD SHAFT.—A wing-fold shaft consists of two interlocking splined sections and two universal joints connected to quill shafts. It provides a telescoping fold joint in the flap drive system linkage between the inboard and outboard wing panels.

Slat System

The slat system, shown in figure 9-37, provides additional lift and stability to the aircraft at lower speeds in the same manner as the leading edge flap

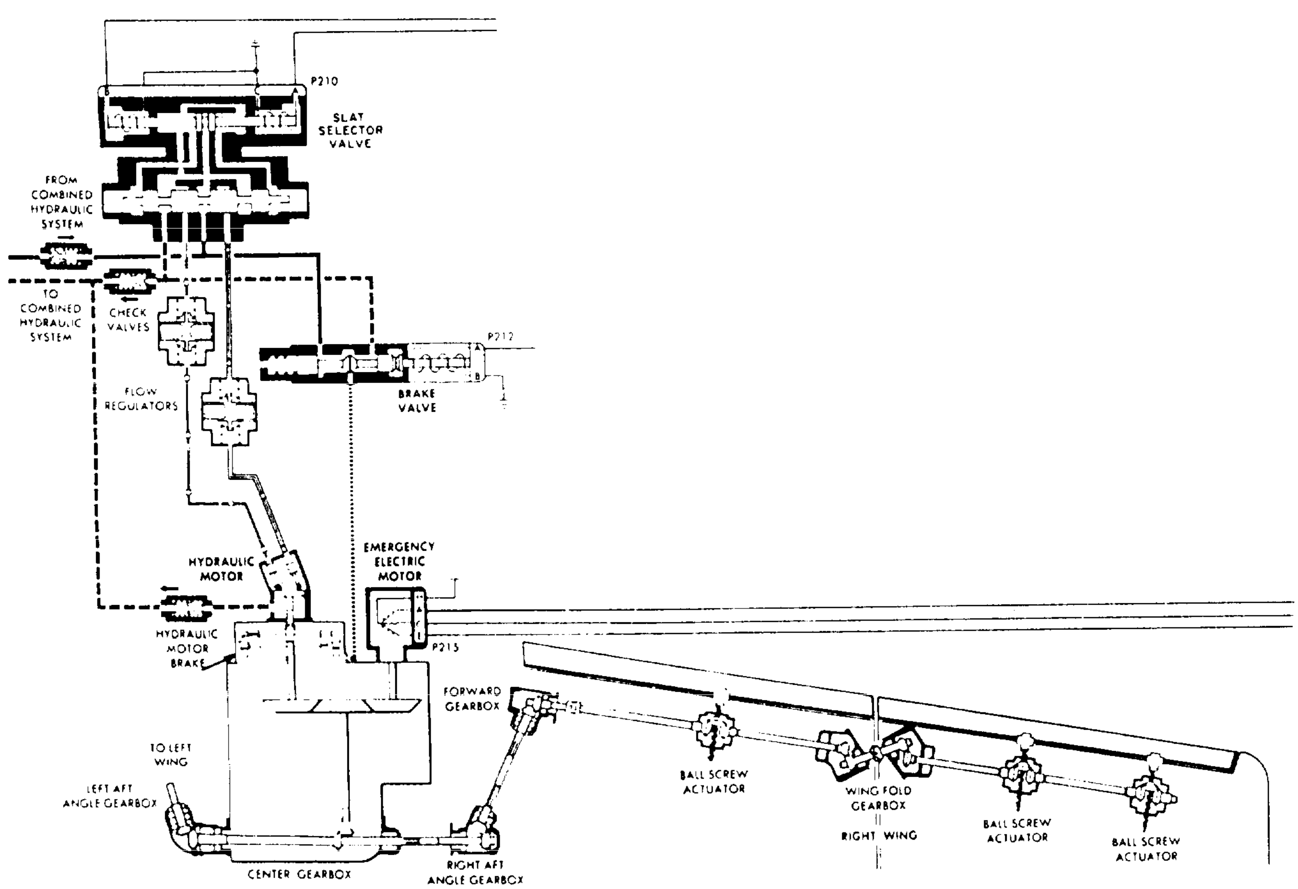


Figure 9-37.—Slat drive system.

system previously discussed. The flap control handle controls the movement of the slats. Moving the flap control handle to the TAKEOFF or LAND position causes the slats to extend to a 27.5-degree leading edge down position.

The slat panels, one inboard and one outboard, interlock by a pin when the wings are spread. When fully retracted, the slats align with the top and bottom wing contours to form the wing leading edge. Shim spacers between the slats and the slat tracks provide adjustment for proper aerodynamic fairing.

Components of the slat system arc similar to those in the flap system. The slats extend and retract by using six series-linked ball screw actuators. The actuators are powered by the hydraulic motor through gearboxes and torque tubes.

If combined hydraulic system pressure fails, the hydraulic motor is locked in the same manner as the flap hydraulic motor, permitting the emergency electric motor to move the slats. Emergency slat operation is accomplished simultaneously with emergency flap operation, using the emergency flap switch. Slat position is also displayed on the cockpit integrated position indicator.

Placing of the flap control handle to either the TAKEOFF or LAND position mechanically closes switches to provide electrical current to the slat selector valve. The selector valve ports hydraulic pressure to the extend side of the high-speed hydraulic motor. This action drives the center gearbox and extends the slats.

Two ball screw actuators drive each outboard slat, and one drives each inboard slat of each wing. Each actuator connects to its downstream actuator by torque tubes and gearboxes. The slats move as one unit. Limit switches in the center drive gearbox de-energize the slat selector valve, blocking flow to the drive motor when the slats fully extend (27.5 degrees) or retract. Placing the flap control handle to the UP position energizes the opposite solenoid of the selector valve and reverses slat motor direction, retracting the slats.

SLAT WING FOLD GEARBOX.—A wing fold gearbox disconnects slat drive linkage at the wing fold joint when the wings fold. The gearbox consists of two identical halves interconnected by a spring-loaded disconnect coupling when the wings are spread. As the disconnect coupling halves move away from each other during the wing folding operation, a spring-operated brake engages, preventing relative motion between the inboard and outboard sections.

SLAT ANGLE GEARBOXES.—Four slat angle gearboxes are provided in the slat system for changing direction of the slat torque tube linkage from the center gearbox to the wing actuators.

DIRECT LIFT CONTROL (DLC)

Direct lift control controls the spoilers and horizontal stabilizers to increase aircraft vertical descent rate during landings. This may be done without changing engine power. Actuating the DLC engage-chaff dispense push-button switch on the control stick grip modifies the pitch and roll computer

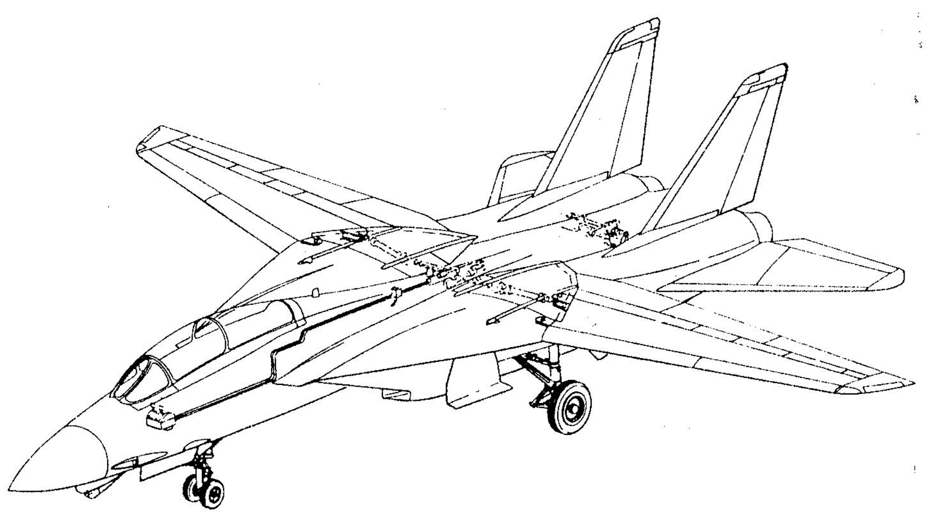


Figure 9-38.—Wing sweep control system.

inputs. This modification causes the eight spoiler actuators to position their spoilers 3 degrees up from the 0-degree position. The pitch computer also generates the DLC servo actuator command drive at the time of DLC engagement. This command drive, which is applied to the DLC servo actuator, drives the stabilizers to the 6-degree trailing edge down position from the 0-degree position. In DLC, the pitch computer and the roll computer permit additional spoiler and stabilizer control through the DLC-maneuver, flap-glove vane thumb wheel control on the control stick grip.

Rotating the thumb wheel fully forward, through modified spoiler and DLC command drives, extends the spoilers to the 12-degree position. The stabilizer is driven to the 8-degree trailing edge down position. Rotating the thumb wheel control fully aft retracts the spoilers to the 4.5-degree position and drives the stabilizers to 0 degrees. This maintains aircraft attitude while changing the vertical descent rate. Direct lift control can be disengaged by momentarily pressing the DLC engage-chaff dispense push-button switch or by setting either throttle lever to military power.

WING SURFACE CONTROL SYSTEM

The wing surface control system controls the variable geometry wings to increase aircraft performance at all speeds and altitudes. The system also provides high lift and drag forces for takeoff and landing. It provides increased lift for maneuvering, and at supersonic speeds, aerodynamic lift to reduce trim drag.

The wing sweep control initiated at the throttle quadrant provides electronic or mechanical control of a hydromechanical system that sweeps the wings. See figure 9-38. The wings sweeps from 20 degrees through 68 degrees in flight. On the ground, a wing sweep position of 75 degrees is available (through mechanical control) for spotting the aircraft or enabling a wing sweep control self-test. See figure 9-39.

Electronic Control

A wing sweep under electronic control is initiated at the throttle quadrant. Four modes are available—automatic, aft manual, forward manual, or bomb manual. Selection of these modes causes the air data computer to generate wing sweep commands consistent with the aircraft speed, altitude, and configuration of the flaps and slats. The commands are applied through the wing-flap glove-vane controller to the wing sweep control drive servo. They are converted to mechanical rotary force. This force, transferred to the wing sweep/flap and slat control box, causes the wing sweep hydraulic control valve to operate hydraulic motors that are driven by the flight and combined hydraulic power systems to sweep the wings. The flight hydraulic power system positions the right wing, and the combined hydraulic power system positions the left wing. A synchronizing shaft (fig. 9-38) interconnects the wings to ensure symmetrical operation. If a hydraulic system fails, it provides the driving force for sweeping the wing affected by the failed system.

Wing sweep commands generated by the air data computer are limited by the configuration of the auxiliary flaps, maneuver flaps, and slats. With the auxiliary flaps extended, wing sweep is limited to 21.25 degrees. The maneuver flaps, with or without slats extension, limit wing sweep to 50 degrees. To prevent structural damage to the wings during negative-g conditions, wing sweep is interrupted to prevent wing sweep changes until the negative-g condition no longer exists. In the automatic mode, the wings are positioned at a rate of 7 degrees per second.

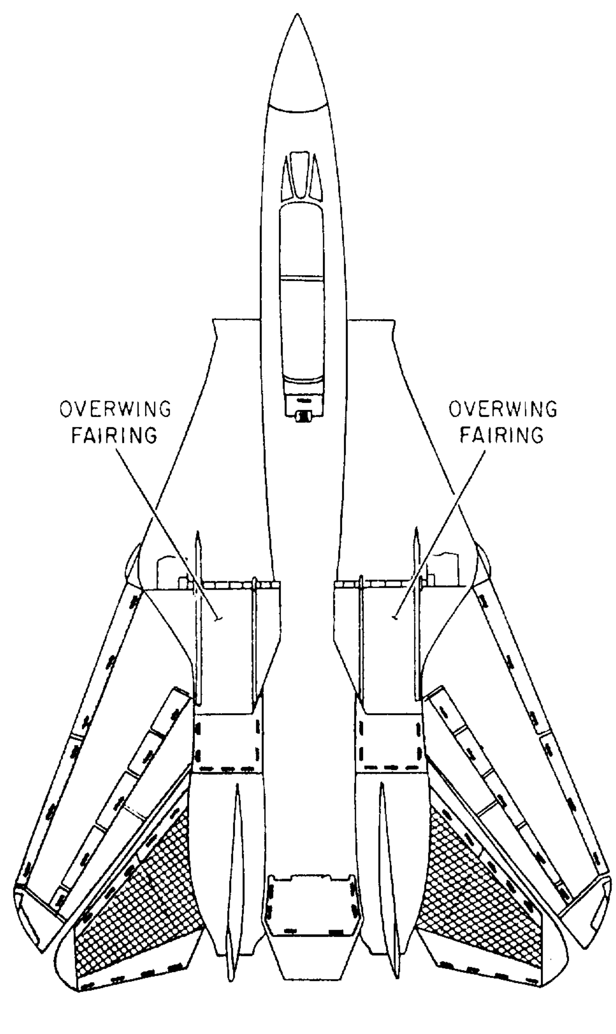


Figure 9-39.—Wing oversweep Position—manual control.

Mechanical Control

When wing sweep is under mechanical control, the wing sweep handle positions the wings through the wing sweep/flap and slat control box. Because the minimum wing sweep limiting is not available under mechanical control, the wings can be swept to an adverse position that could cause damage to the wings. Mechanical control is used for emergency wing sweep and wing oversweep.

During emergency wing sweep, the wing sweep handle, mechanically coupled to the wing sweep/flap and slat control box through a cable assembly, positions the wings. The wing sweep can be returned to electronic control by repositioning the wing sweep handle to the stowed position.

Wing oversweep can only be obtained with the aircraft weight on the wheels. Wing oversweep, shown in figure 9-39, reduces the amount of space required for spotting the aircraft. A wing sweep self-test can only be performed while the wings are overswept.

SPEED BRAKE SYSTEM

Speed brakes are hinged, movable secondary control surfaces used for slowing down the speed of the aircraft by increasing the profile drag. These surfaces are also called “dive brakes” or “dive flaps.” On some aircraft, they are hinged to and faired with the side or bottom of the fuselage. On others, they are attached to the wings. Regardless of their location, their purpose is the same.

Fuselage Type

The fuselage speed brake system is normally electrically controlled and hydraulically operated. See figure 9-40. In an emergency, it can be controlled manually.

The brake surfaces are installed on the sides of the aft portion of the fuselage below and forward of the horizontal stabilizer. They hinge at their forward end. When in the closed position, they fit flush with fuselage skin. An elevator speed brake interconnect provides a connection between the left-hand speed brake and the aircraft nose down elevator control cable. When the speed brakes open, the cable pulls and provides a nose down action to counteract the tendency of the aircraft to assume a nose up condition.

The speed brakes may be actuated by the two-position, spring-loaded-to-neutral control switch on the throttle lever or by the manual override control handle. When operating the switch to open the speed

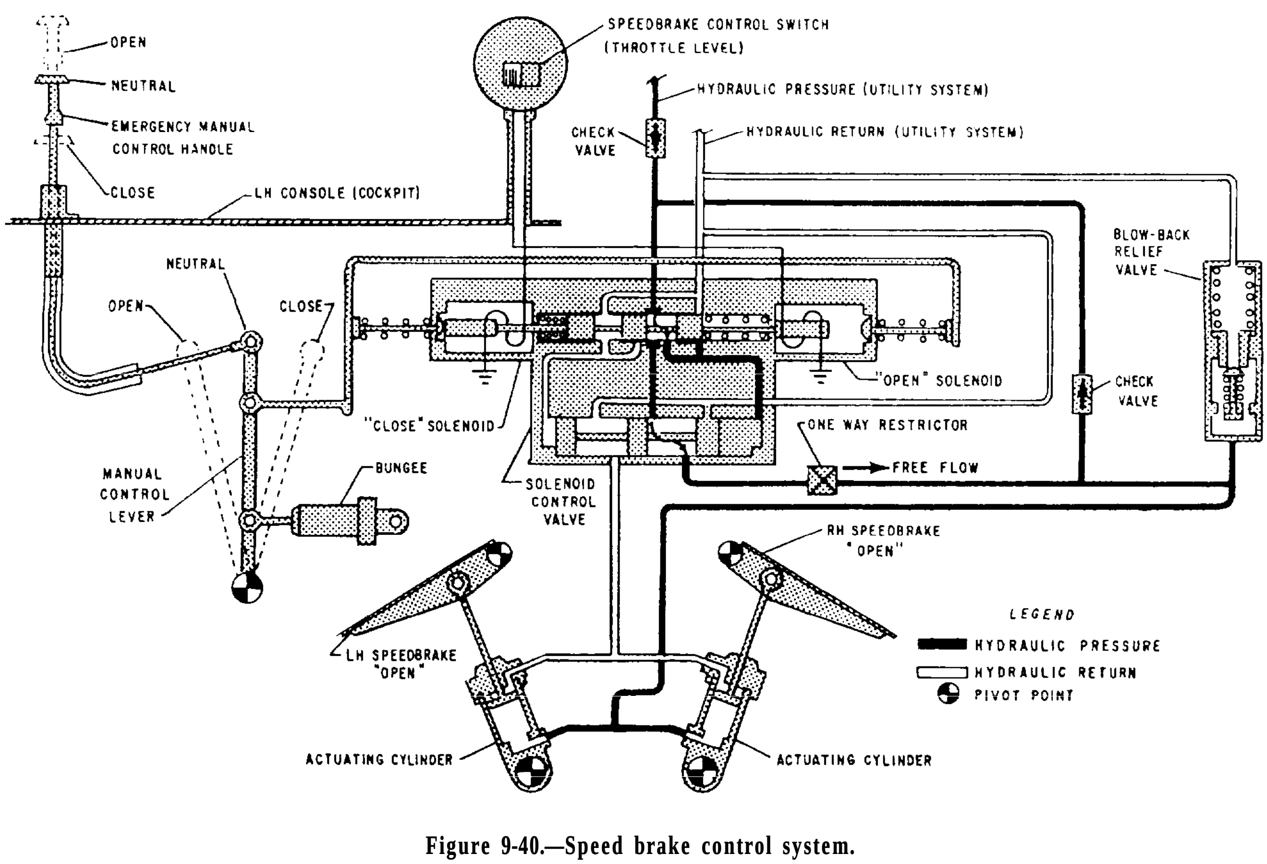


Figure 9-40.—Speed brake control system.

brakes, the control circuit energizes to operate the opening solenoid of the control valve. Pressure is sent to the actuating cylinders, extending the speed brakes. To close, the opposite solenoid energizes, repositioning the control valve and directing pressure to the retract side of the actuating cylinders, closing the speed brakes.

When you depress or pull the manual override handle to operate the speed brakes, a plunger manually positions the control valve to direct pressure to the actuating cylinders. The spring bungee connected to the manual control lever returns the manual override handle assembly to the neutral position when the handle is released. If electrical power is applied while the manual override handle is actuated, the system will remain in the position selected by the handle. If the handle is released, the system actuates to the position selected by the control switch on the throttle lever. The speed brakes cannot be stopped at intermediate positions between fully closed and fully open. The restrictor in the open line restricts return fluid flow from the actuating cylinders when the speed brakes are being closed.

If the hydraulic system fails, the check valve in the pressure line traps pressure between the control valve and the actuating cylinders. If the speed brakes are open, this pressure will hold them open. If the speed brakes are actuated to the closed position, the pressure in the system will shift the primary slide in the control valve. This movement will relieve the trapped pressure and allow the speed brakes to close from the air load against them.

A blowback relief valve, installed in the hydraulic return line, allows for automatic retraction of the speed brakes under high air loads. When the speed brakes are open, the force of the airstream against the surfaces tends to force them closed. The force builds up the hydraulic pressure in the speed brake system. When the pressure reaches a maximum of 3,650 psi, it relieves through the blowback relief valve.

Wingtip Type

The wingtip speed brake system is an electrically controlled and hydraulically operated system. It operates either alone or with the fuselage speed brakes. See figure 9-41.

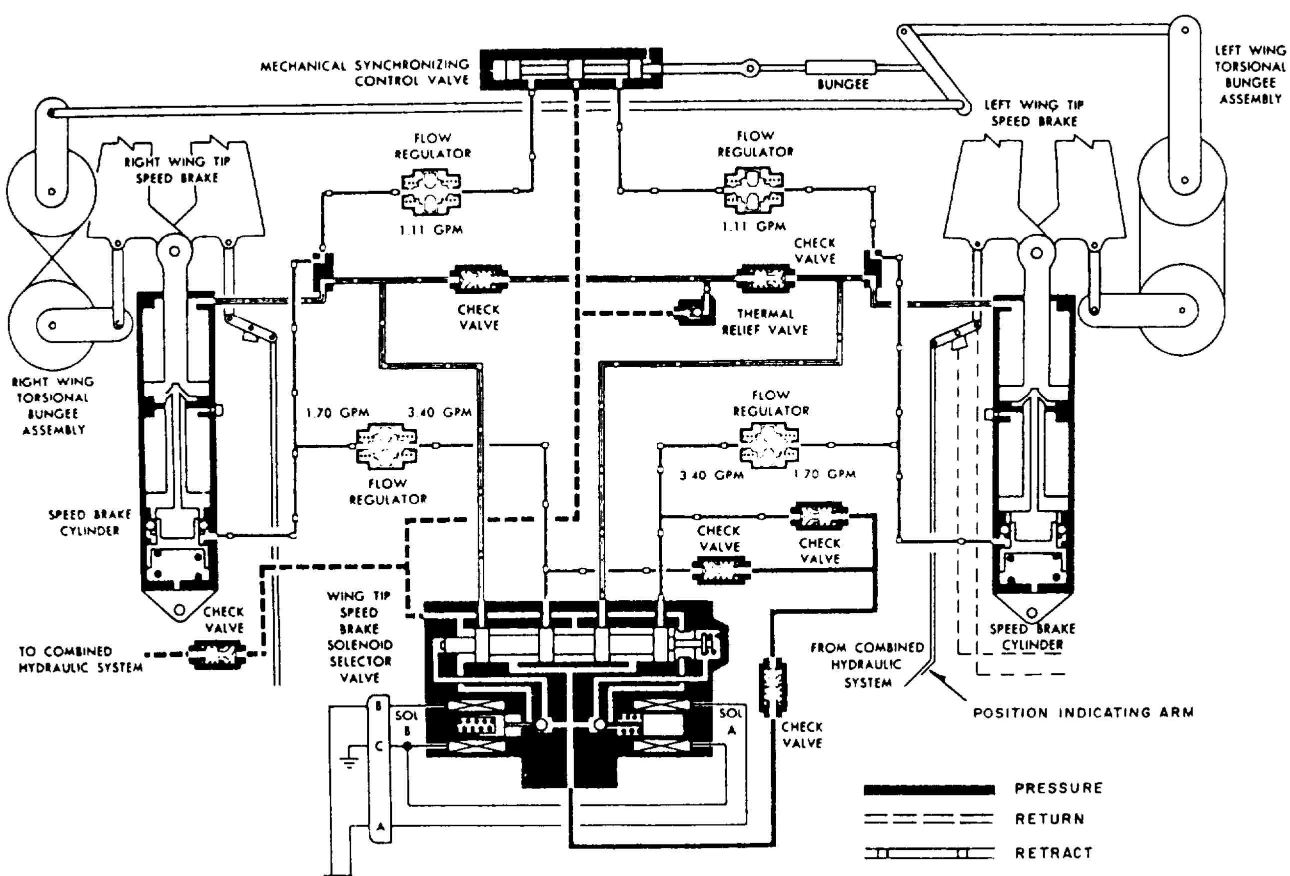


Figure 9-41.—Wingtip speed brake control system.

The wingtip brake consists of a set of trailing edge surfaces for each wing. The lower half attaches to the wing structure with two external fixed hinges. The upper half is attached to the wing at the same wing station with two adjustable tension lengths. An interconnecting hinge between the upper and lower halves provides a common connection point for the actuating cylinders. The hinge provides symmetrical deflection of upper and lower panels. Each panel can open up to 60 degrees for a total angle of 120 degrees for each wingtip brake. When retracted, they lie flush with the wing surface. They can extend and hold at any angle between 0 and 60 degrees, depending upon the amount of aerodynamic braking desired.

A mode selector switch permits simultaneous or independent operation of the wingtip and fuselage speed brakes, with the speed brake control switch located on the right throttle quadrant power lever. Moving the SPD BRK switch to the forward position closes the brakes. Moving it to center position holds the brakes at any desired angle. Moving it aft opens the brakes. The switch is spring-loaded to neutral from the aft position only.

Selecting the open position energizes the selector valve, porting hydraulic pressure from the combined hydraulic system to the extended side of the actuating cylinder. When the switch is positioned to closed, the opposite solenoid energizes. Pressure is ported to the retract side of the actuating cylinders. With the switch in neutral, hydraulic fluid is blocked from both the extend and retract sides of the speed brake cylinders. This action hydraulically locks the speed brakes. If the electrical circuit fails, the selector valve is de-energized as a fail-safe feature and the speed brakes retracts.

The wingtip speed brake control system normally depends upon the hydraulic flow regulators to maintain symmetrical extension of the left and right brakes. If a malfunction causes asymmetry of extension, an electrical disparity signal is sensed by the speed brake null detector. When the disparity between the extension of the left and right brake reaches 8 degrees, the null detector de-energizes the selector valves and causes the speed brakes to close.

On some aircraft, the synchronization mechanism (fig. 9-41) consists basically of synchronizing linkage, two torsional bungee assemblies, and a cable run interconnecting the three mechanisms to a mechanical synchronizing control valve. The synchronizing mechanism is a comparative linkage type that senses unequal motion between the two brake surfaces. Movement of either speed brake transmits through the torsional bungee assembly and the cables to the synchronizing mechanism. Any unequal movement upsets the synchronizing mechanism's neutral position, displacing the synchronizing valve shuttle. When the speed brakes are opening or closing, the valve is normally in neutral as long as the travel of both sides is equal. When unequal travel moves the valve shuttle out of neutral, the valve will relieve hydraulic pressure from the speed brake actuating cylinder, producing the hugest opening angle. This decelerates the opening of the speed brake or bleeds down the speed brake with the largest angle until the disparity is within limits and the shuttle returns to neutral. On later models this mechanical synchronization system has been deleted.

If the mechanical synchronization system fails to maintain synchronization within 8 degrees, the electrical fail-safe system operates and de-energizes the selector valve to close the speed brakes. If the synchronizing linkage becomes jammed, the torsional bungee assembly can be forced out of detent, isolating the linkage from the speed brake and preventing damage to the linkage because of overloads.

The bungee in the synchronizing mechanism linkage acts as a rigid length to the synchronizing valve during normal operation of the wingtip speed brake. If the valve becomes jammed, abnormal loads on the bungee will cause it to give and relieve the excessive loads before damage to the valve, linkage, or bungee occurs.

TRIM SYSTEM

A trim system is provided in the flight controls to lessen the need for constant effort on the part of the pilot to maintain the desired heading and altitude. The trim system stabilizes the aircraft during flight.

Lateral Trim

The aileron trim control system is shown in figure 9-42. The illustration represents a trim tab arrangement similar to that found on aircraft equipped with conventional aileron systems.

Operation of the lateral and longitudinal trim systems is usually controlled by a five-position, four-throw, momentary ON contact switch with a center OFF position. The switch is found on the control stick grip. This switch electrically energizes the trim control motor, which operates the trim control actuator to reposition the load-feel bungee and achieve hydraulic-powered actuation of the ailerons. At the same time, the actuator operates the cable drum mechanism. The cable drum mechanism operates the jack screw mechanism to reposition the follow-up trim tab to aerodynamically maintain the aileron surface in a position corresponding to that achieved by the hydraulic actuation.

The tab movement does not control the lateral trim of the aircraft while normal powered flight is maintained. This is accomplished by the hydraulic-powered displacement of the ailerons. When the manual flight control system is used, the follow-up trim tab position introduced during powered operation becomes effective and maintains the same trim as that provided by the powered operation.

With the power system disconnected, further hydraulic trim control ends, and all future trim inputs are achieved through aerodynamic effect. This function depends upon selective follow-up tab position. Engaging the AFCS controls the trim actuator by electrical inputs.

Aircraft without trim tabs achieve lateral trim by repositioning the lateral control surfaces as necessary to achieve a balanced lateral flight condition. The trim actuator, located in the aileron trim and mixing linkage, normally acts as a series-connected, fixed-length rod in the aileron control system. The trim control switch on the stick grip controls the actuator length. Shortening or retracting the trim actuator (trim button to the right) supplies a left wing up input into the aileron control system linkage. Extending the actuator supplies a left wing down input. The trim actuator changes the neutral position of the aileron mechanism, allowing the control surfaces to deflect and trim the aircraft without moving the control stick.

Longitudinal Trim

Longitudinal or pitch trim can be accomplished in several ways. On aircraft with a nonmoveable horizontal stabilizer, trim could be provided by a trim tab arrangement or deflection of the elevators in much the same manner as described for the lateral trim systems.

Aircraft with a movable horizontal stabilizer and elevators are longitudinally trimmed by changing the angle of incidence of the stabilizer. Moving the four-way trim control switch on the stick grip fore or aft will raise or lower the leading edge of the stabilizer to provide the angle of incidence necessary for balanced flight. An electric trim motor and actuator arrangement provides movement of the stabilizer.

Aircraft that use a movable horizontal stabilizer for longitudinal control trim do so by varying the neutral position of the control linkage, which, in turn, moves the surface. For example, longitudinal trim is provided by varying the position of the artificial-feel bungee, repositioning the linkage, and setting up a new neutral position for the stabilizer linkage. Anytime a new neutral is introduced by the trim actuator, the power valve shuttle is displaced. The stabilizer assumes a new neutral location, changing

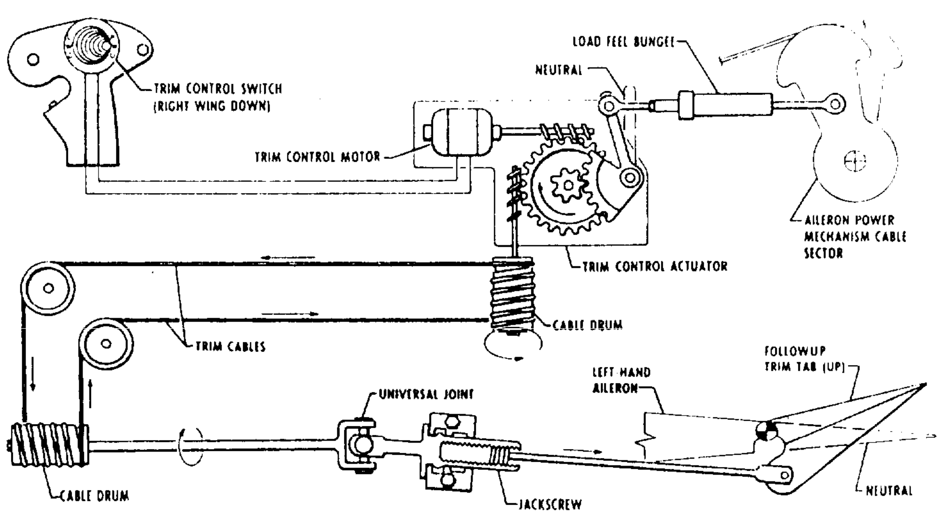


Figure 9-42.—Aileron trim control system.

the attitude of the aircraft. The trim inputs may be provided by the pilot or the automatic flight control system. The actuator has two operating speeds-high speed for manual trim and low speed for AFCS trim.

Directional Trim

Directional trim is necessary to compensate for yaw of the aircraft. Rudder trim is basically similar to the aileron trim. When the momentary throw rudder trim switch moves left or right, the trim actuator energizes to move the load-feel bungee, repositioning the rudder power mechanism input crank. The rudder linkage and the rudder are repositioned accordingly to a new neutral position.

Most aircraft with power-controlled actuators work in a similar manner, using an electric trim actuator to change the neutral position of linkage, deflecting the rudder to maintain the desired directional stability. Like the lateral and longitudinal trim systems, rudder trim action can be accomplished manually or automatically. Trim position indicators provide a cockpit indication of the amount of trim or surface deflection required by each trim system.

CONTROL SYSTEM MAINTENANCE

Organizational maintenance of the secondary flight control system includes checking system operation, rigging, periodic inspection, lubrication, isolation of malfunctions, and replacement of faulty components.

Proper operation of the gearboxes, interconnecting splined shafts, and screw jack actuators are dependent on proper lubrication. Lack of proper lubrication will generally result in binding and excessive loading of torque tube assemblies. Lack of proper lubrication promotes corrosion. Space and time limitations during shipboard operations often detract from the timely access to some of the slat and flap actuators. In many cases a wing spread and extension of the surfaces are necessary. Attention to these corrosion-prone areas will materially contribute to trouble-free operation of the screw jack mechanisms.

Repair of most of the gearboxes and screw jack actuators at the intermediate level of maintenance is limited to replacement of nuts, bolts, washers, gaskets, bearings, and shims. At the intermediate level of maintenance, components of a secondary flight control system may be disassembled for routine maintenance, such as cure date seal and miscellaneous parts replacement.

> **NOTE:** Before disassembly of any component, reference should be made to the "Intermediate Maintenance" section of the applicable MIM or accessories manual to determine repair procedures and test equipment requirements. If the component is beyond the repair capability of a given activity, it should be forwarded through channels to an authorized higher level repair activity.

The repair process for many of the flap hydraulic components will generally include the following considerations:

1. Clean the disassembled part, using a suitable solvent followed by air drying with low-pressure air.

2. Inspect all parts, using a strong light and some means of magnification, or one of the nondestructive methods of metal inspection. Threaded parts are inspected for crossed, stripped, worn, or otherwise damaged threads. Springs are checked for distortion, permanent set, and alignment. Spring alignment may be verified by rolling them on a smooth, flat surface. The free length, compressed length, and reflected load of the springs should be verified in accordance with the values provided in the applicable MIM.

3. Inspect mated surfaces for excessive wear, separation of plating, and evidence of nicks or scratches. All parts that show signs of excessive scoring, pitting, or other surface irregularities should be replaced. Minor imperfections can sometimes be removed with fine crocus cloth or lapping compound, depending on the design and tolerance specifications of the part.

4. Be sure that all passages and chambers of the part under repair are clean and free from obstructions.

> **NOTE:** During the complete repair process, cleanliness of the work area, as well as the external and internal parts, is a prime consideration. The close tolerance mated surfaces within most hydraulic components are extremely susceptible to damage by contamination regardless of the manner of introduction.

Following reassembly, the component must be bench tested to verify its proper performance. Usually, testing will include proof testing, leakage testing to verify proper internal seal operation, and operational testing.

Quality assurance verification is required throughout the repair process and at the completion of repair. All repairs must be accomplished as specified in the "Intermediate Maintenance" section of the applicable MIM or 03 accessories manuals. Steps that require quality assurance verification are so indicated by appearing in italics, being underlined, or some other obvious manner. Following repair, partially fill the component with preservative hydraulic fluid and cap and/or plug to prevent contamination.

MAJOR ASSEMBLY REMOVAL/INSTALLATION AND AIRCRAFT ALIGNMENT

Learning Objective: *Recognize the procedures for the removal and installation of wings, stabilizers, and flight control surfaces, and the subsequent alignment checks.*

The primary flight control surfaces and some of the secondary control surfaces are attached to the wings and stabilizers of the aircraft. In many instances, the wings and stabilizers are damaged beyond repair. When this occurs, the wings and stabilizers must be removed and sent to a depot-level maintenance facility for repair, and a replacement installed.

WINGS

Removal and installation of a wing are major operations that require experienced personnel and close supervision by a senior petty officer.

You should read the airframes section of the applicable MIM carefully before attempting to remove a wing. This manual will give step-by-step instructions for wing removal and installation. It is necessary to follow these instructions to prevent possible damage caused by failure to disconnect or connect units in the proper sequence.

Listed below are some general precautions that you should observe when removing and installing a wing or wing section.

1. The aircraft should be placed in a hangar or other area protected from the wind.

2. Make certain all the necessary equipment is available and at hand. A list of the necessary special tools and equipment can be found in the applicable MIM.

3. Ensure that you have sufficient manpower for proper handling.

4. Ensure that all screws, bolts, and other removed fasteners are placed in containers and properly marked to prevent loss.

5. Ensure that all removed fairings are marked and stowed in a safe place.

6. In disconnecting tubing, electrical connectors, control cables, and bonding wires, see that the instructions given in the aircraft MIM are carried out.

7. Make certain that all disconnected tubing is capped.

8. If hoisting equipment is to be used, be sure it is in good condition and a qualified operator is available. Also, ensure the hoist fittings are properly installed. Some wings will not balance at their hoist fittings, which makes it necessary to attach guide ropes to keep the wing steady after it is disconnected from the aircraft.

9. Before attempting to remove any structural bolts, make certain that the wing is properly supported with all loads removed from the fittings. A mallet and brass drift pin may be used in removing these bolts.

10. After the wing is removed from the aircraft, all fittings, connections, and unremoved structural members should be inspected for secondary damage before installing the new wing or wing section. (Secondary damage is damage to adjacent structures, which may have resulted from the transmission of the shock or load that caused the primary damage.)

11. Before installing the new wing, you should take advantage of improved accessibility to inspect and repair corrosion damage, and renew preservative coatings in previously inaccessible areas.

The petty officer in charge should ensure that the following general precautions are taken in installing a wing or wing section.

1. Check the identification tag of the new assembly to make sure it is the correct replacement unit.

2. See that extreme care is taken in removing the wing from the container, preventing any possible damage.

3. Inspect the new wing or wing section for possible damage incurred during shipment or removal from the container. The container should be used for shipping the damaged wing to the depot maintenance facility.

4. With the wing in position for installation and properly supported, ensure that all structural bolts are installed and the nuts properly torqued.

CAUTION

The attaching bolts should never be forced; if they bind, check alignment of the wing. Forcing the attaching bolts will result in damage to the wing structure.

5. Make certain that all tubing, electrical connectors, control cables, and any other disconnected mechanisms are properly connected.

6. Check the operation of all mechanisms that were disconnected during removal. Make the necessary rigging adjustments in accordance with the applicable MIM before installing access doors and fairings.

7. Make a final inspection of the completed job.

STABILIZERS

The removal and installation of stabilizers are similar, in most cases, to that of wings and wing panels. On many aircraft the horizontal stabilizer is a movable airfoil, controllable from the cockpit. On some of these aircraft, it is used in conjunction with the elevators to maintain longitudinal control at sonic speeds where the elevators have a tendency to lose their effectiveness. On other aircraft the movable horizontal stabilizer serves the dual purpose of elevators and stabilizers and, in many instances, is referred to as a stabilator.

Some aircraft have an empennage or tail group that consists of all-movable horizontal stabilizers and a single all-movable vertical stabilizer. These aircraft do not have elevators or a rudder.

The removal and installation of stabilizers, like that of the wing, are major jobs and must be accomplished with care and close supervision. Step-by-step instructions of the removal and installation of stabilizers are also included in the "Airframes" section of the applicable MIM. Many of the general precautions listed under "Removal and Installation of Wings" also apply to stabilizer removal and installation.

FLIGHT CONTROL SURFACES

It is sometimes necessary to remove control surfaces from aircraft to repair or replace them. The instructions presented in the following paragraphs are general instructions, applicable to several types of aircraft. For specific instructions and precautions, you should always consult the MIM before removing a control surface from any aircraft.

Removal of a control surface should not be attempted until the aircraft is placed in a hangar or an area protected from the wind. Before any control surface is removed from the aircraft, it should be tagged with the bureau number of the aircraft and the location of the control surface on the aircraft.

The first step is to remove the access covers and fairings. To prevent the loss of these parts, they should be left attached to the aircraft by one screw or by a piece of safety wire. The other screws should be put in a container to prevent them from being lost.

Disconnect bonding wires, electrical connectors, and control linkage. Before disconnecting cable linkage, you should relieve the tension at the most convenient turnbuckle. Next, support the entire control surface, either manually or with mechanical supports, in such a manner as to remove all the load from the hinges. Remove the hinge bolts by using a mallet and brass pin. The control surface should be supported and all the hinges kept in alignment until the last hinge bolt has been removed. On long controI surfaces, it may be necessary to replace the hinge bolts with drift pins to keep the hinges aligned while removing the remaining hinge bolts.

Control surfaces are sometimes attached with piano wire hinges. Removal of the piano wire can be accomplished by removing the ends, securing one end of the wire in the chuck of a hand drill, and rotating the wire with the drill while withdrawing it. Excessive spinning will have a wearing effect on the hinge material and should be avoided. The reuse of piano hinge wire is not safe; therefore, any wire removed should be discarded.

After all the hinges are disconnected, remove the control surface from the aircraft and support it carefully to prevent damage to the hinge brackets and adjoining surfaces. Replace the hinge bolts in the hinges to prevent them from being lost or damaged.

Before installing a control surface, check the identification tag to determine its proper location on the aircraft. Place the surface in position carefully. You should ensure that all the hinge holes are properly aligned. Drift pins may be used to align the holes. With the control surface correctly supported, install the hinge bolts. For a surface attached by piano hinge wire, a new wire should be used. After a control surface is installed, connect the control linkage and check the rigging of the system.

Some flight control surfaces are balanced at the time of manufacture by adding counterweights to the inside of the leading edge of the control surface. This balance must be maintained (within certain tolerances) throughout the service life of the control surface because flutter or dynamic oscillation of these surfaces in flight is not desirable. Balance tolerances are always specified in the aircraft structural repair manual.

Alignment of the airframe structure means checking the position relationship of each major component—the wing group, tail group, and fuselage group-to the other. The alignment of the airframe is important since it is directly related to the aerodynamic performance of the aircraft. Misalignment may affect the flight characteristics of the aircraft, and consequently, the efficiency of the pilot-aircraft combination.

For this reason and for purposes of determining if any hidden structural failures exist, an alignment check should be performed when an aircraft has encountered excessive g's in flight, when a hard landing has been experienced, or when the aircraft has been subjected to extensive damage.

The need for an alignment check after extensive damage is rather apparent; however, this is not necessarily so in situations where the aircraft exceeds the g design limit or where a hard landing has been experienced. The alignment check under these conditions may expose damage that might otherwise go unnoticed.

ALIGNMENT LEVELING METHODS

Prior to making an alignment check, it is necessary to level the aircraft both laterally and longitudinally. This may be accomplished by using the transit, spirit level, or plumb bob and datum plate method. You should always use the method of leveling specified by the manufacturer.

When you are leveling an aircraft for an alignment check, the aircraft should be inside a hangar where air currents will not interfere with the accuracy of the alignment readings. Jacks should be used to control the attitude of the aircraft during the check.

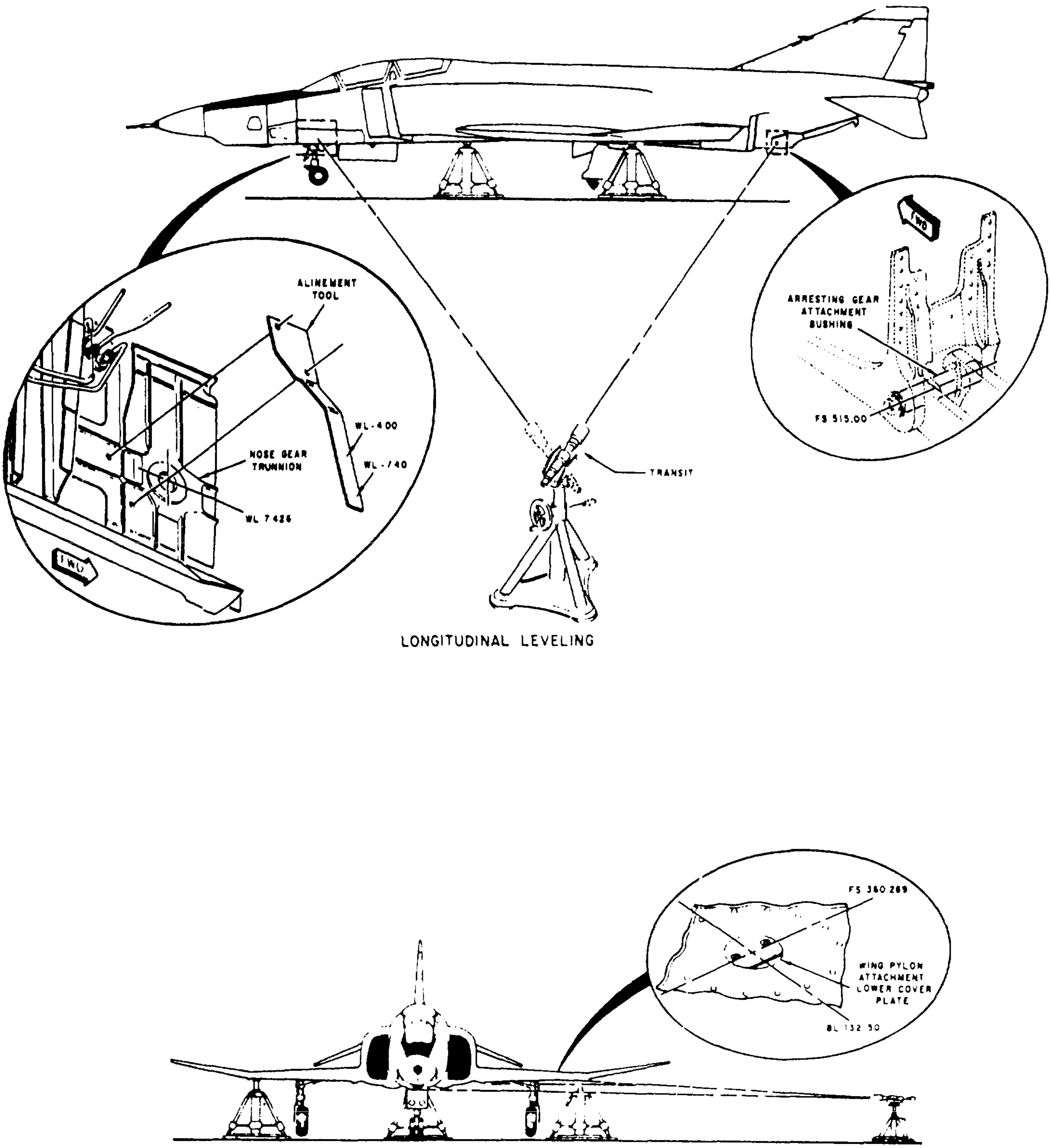


Figure 9-43.—Transit leveling.

Transit

The transit method is the most accurate. Transit leveling is accomplished by sighting specified points on the aircraft. Two longitudinal and two lateral points are used for this method. The reference points are sighted through a surveyor's transit. Figure 9-43 illustrates longitudinal and lateral leveling of an aircraft using the transit method.

Spirit Level

Aircraft that use the spirit level method have leveling lugs either built into the structure or provisions for mounting them on the structure. The leveling lugs are usually in the nosewheel well. Spirit leveling lugs are shown in figure 9-44.

> **NOTE:** The leveling lugs should be inspected for possible damage or misalignment prior to leveling the aircraft. In the event of damage to the leveling lugs, the repaired lugs must be calibrated by cross-reference with the transit leveling method.

Plumb Bob and Datum Plate

This method uses a datum plate or scale mounted on the deck of a compartment. Provisions for hanging

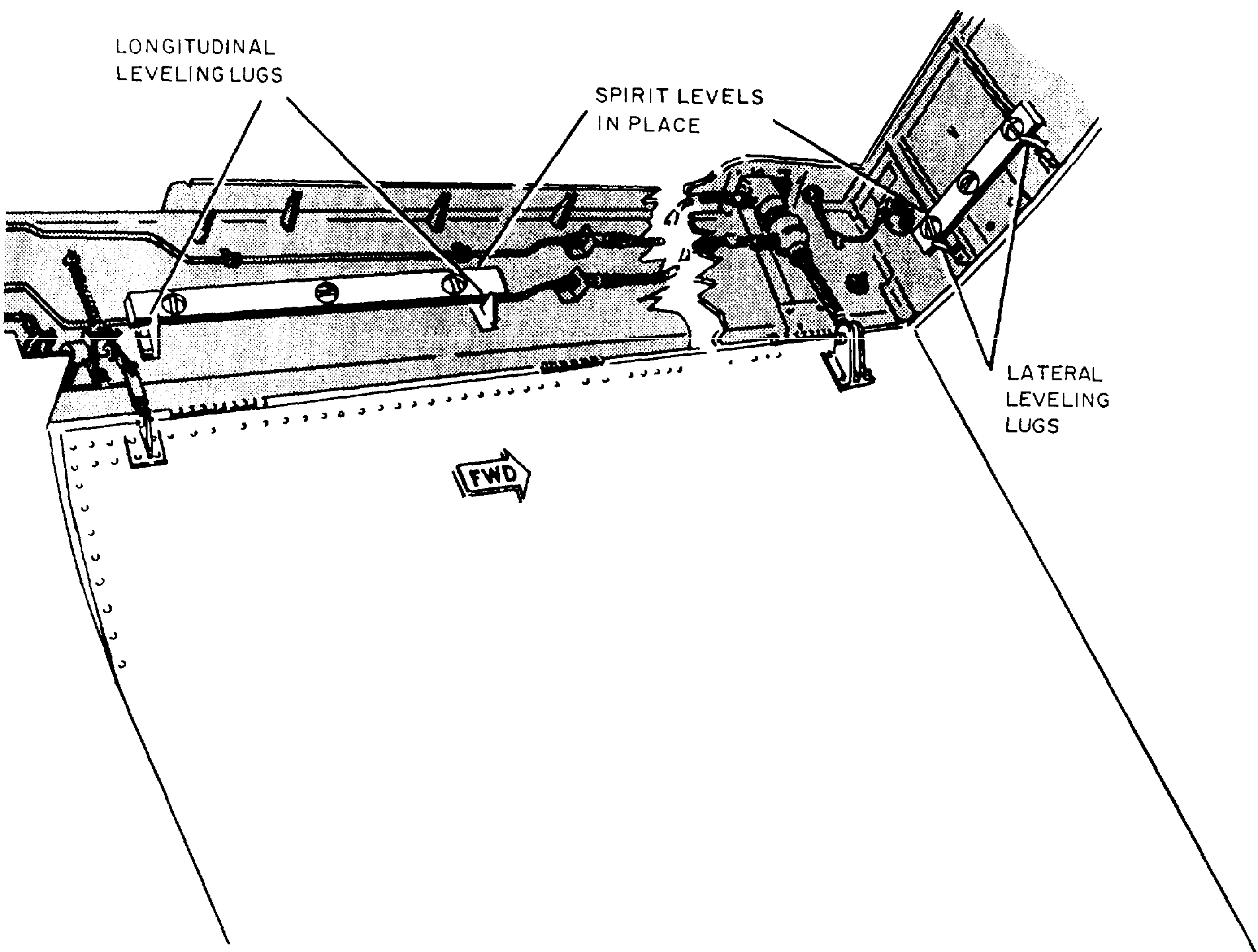


Figure 9-44.—Spirit leveling.

the plumb bob are located directly above the datum plate. The aircraft is level when the plumb bob pointer is at 0 degrees on the datum plate. Figure 9-45 shows the plumb bob and datum plate method of aircraft leveling.

ALIGNMENT CHECK

The alignment or symmetry check is made after the aircraft has been leveled. This check is made by measuring the distance between certain points on the aircraft. These points are selected because they are relatively static and because their location will best reflect any misalignment. Most manufacturers recommend that the measurements be taken directly from one specified point to another. Figures 9-46 and 9-47 show alignment checks.

On other types of aircraft, drop points are provided at various locations for use in checking the alignment. Plumb bobs are dropped from each of these points to the reference plane (floor) so that the pattern for measurement may be described. When you are using this method, the elevation check dimensions are measured from the drop points to the reference plane; in this case, the floor. The horizontal check dimensions are measured from one point (described by the plumb bob), along the reference plane (floor), to another point.

If the alignment check measurements exceed the tolerances listed in the aircraft structural repair manual, the aircraft must be considered non-airworthy until a special disposition can be made by higher authority.

WING TWIST CHECK

With the aircraft leveled and the wings folded, it is possible to check the wings for twist. One checkpoint is provided on each wing. Clinometer readings taken at these points, when compared to the fuselage longitudinal clinometer readings, will enable you to determine the condition of each wing. This is possible because there is a definite relationship between the fuselage longitudinal and wing reference

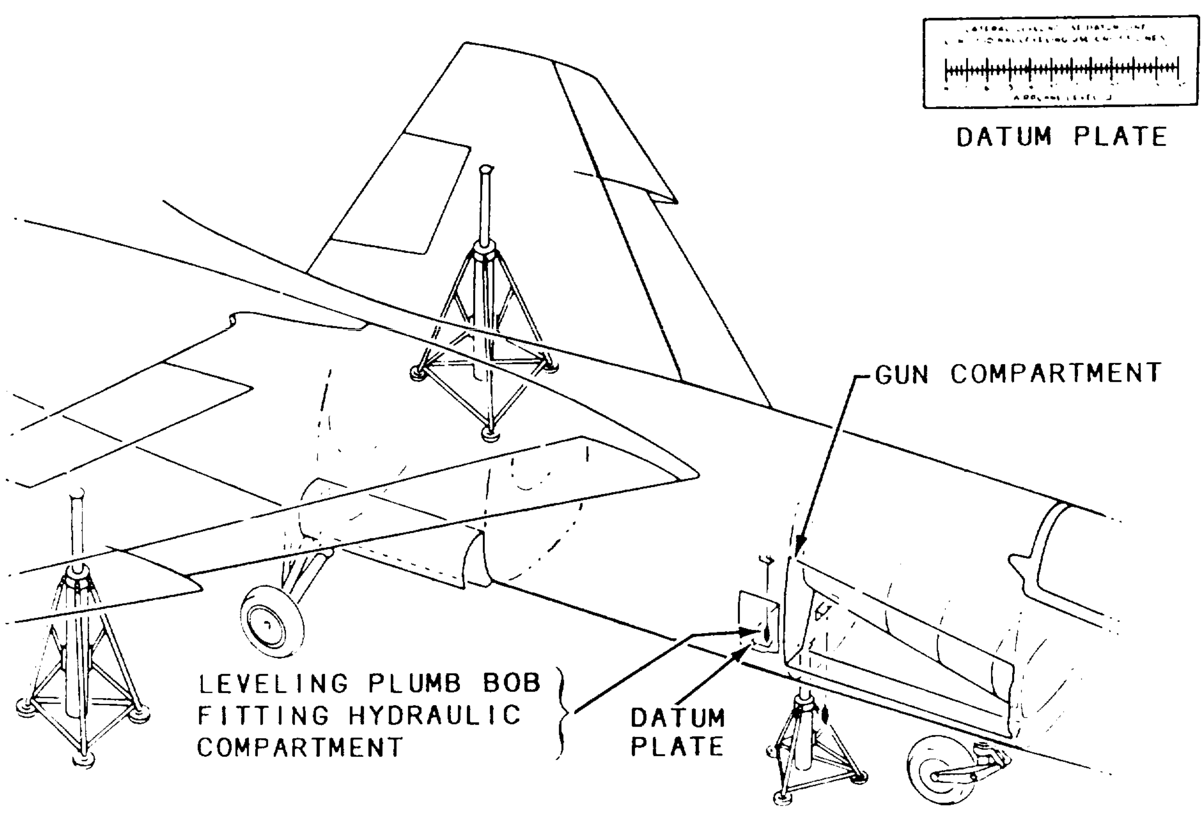


Figure 9-45.—Plumb bob and datum plate leveling.

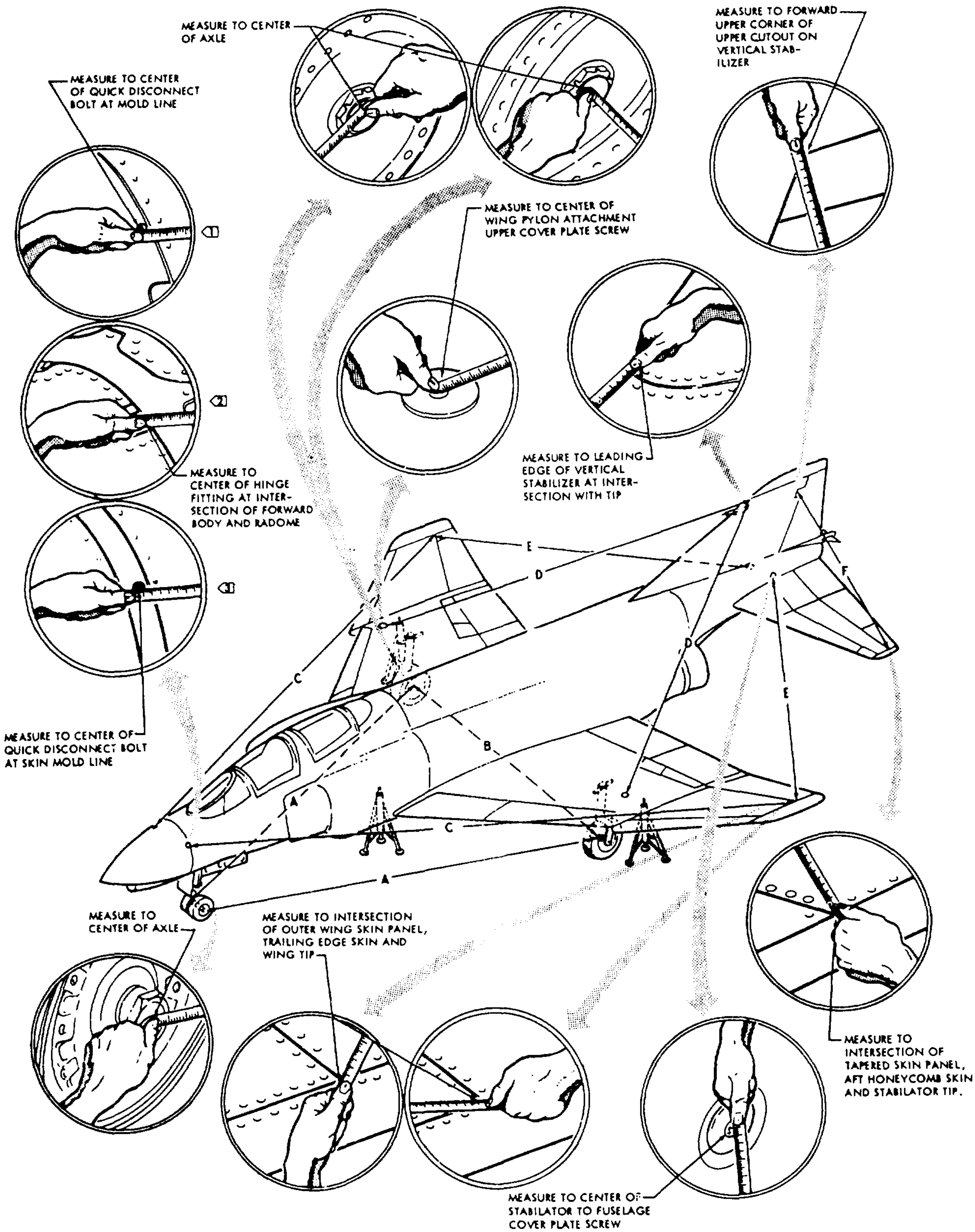
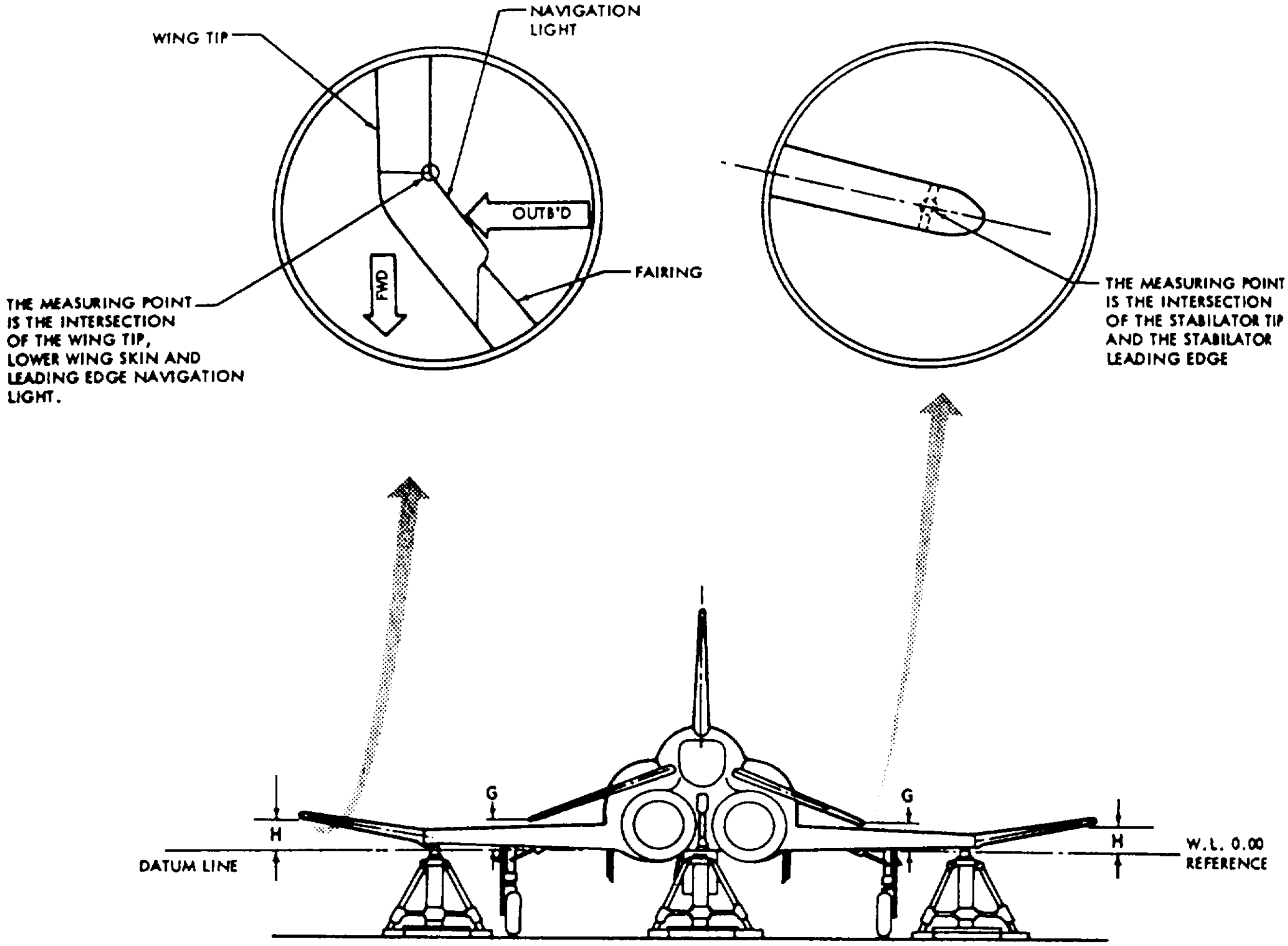


Figure 9-46.—Aircraft alignment data.



REF. LETTER	DIMENSIONS		DIMENSION TOLERANCE BETWEEN AIRCRAFT		DIMENSION TOLERANCE BETWEEN LEFT & RIGHT SIDE OF AIRCRAFT	
	INCHES	FEET & INCHES	INCHES	INCHES	INCHES	INCHES
A	294.00	24' 6"	± 0.75	± 3/4	0.75	3/4
B	206.46	17' 2 7/16"	± 1.00	± 1		
C	476.45	39' 8 7/16"	± 1.00	± 1	0.50	1/2
D	292.05	24' 4 1/16"	± 0.75	± 3/4	0.375	3/8
E	260.08	21' 8 1/16"	± 1.00	± 1	0.50	1/2
F	143.15	11' 11 1/8"	±0.50	± 1/2	0.375	3/8
G	17.403	1' 5 3/8"	± 0.375	± 3/8	0.375	3/8
H	19.66	1' 7 11/16"	± 0.375	± 3/8	0.375	3/8

Figure 9-47.-Aircraft alignment data and measurement chart.

lines. You should follow the following steps to perform a wing twist check:

1. Fold the wings and level the aircraft laterally.

2. Install the leveling bar in the forward lockpin holes of the outboard panel fold rib.

3. Turn the rod until the milled flat at the forward end is straight up.

4. Set the clinometer on the flat and record the reading when the dial has stopped rotating.

The right- and left-hand wing readings must be within 0 degrees, 12 minutes of each other for acceptable aerodynamic tolerances with respect to twist. They must also fall within the following upper and lower limits. The lower limit is established by subtracting 0 degrees, 20 minutes from the

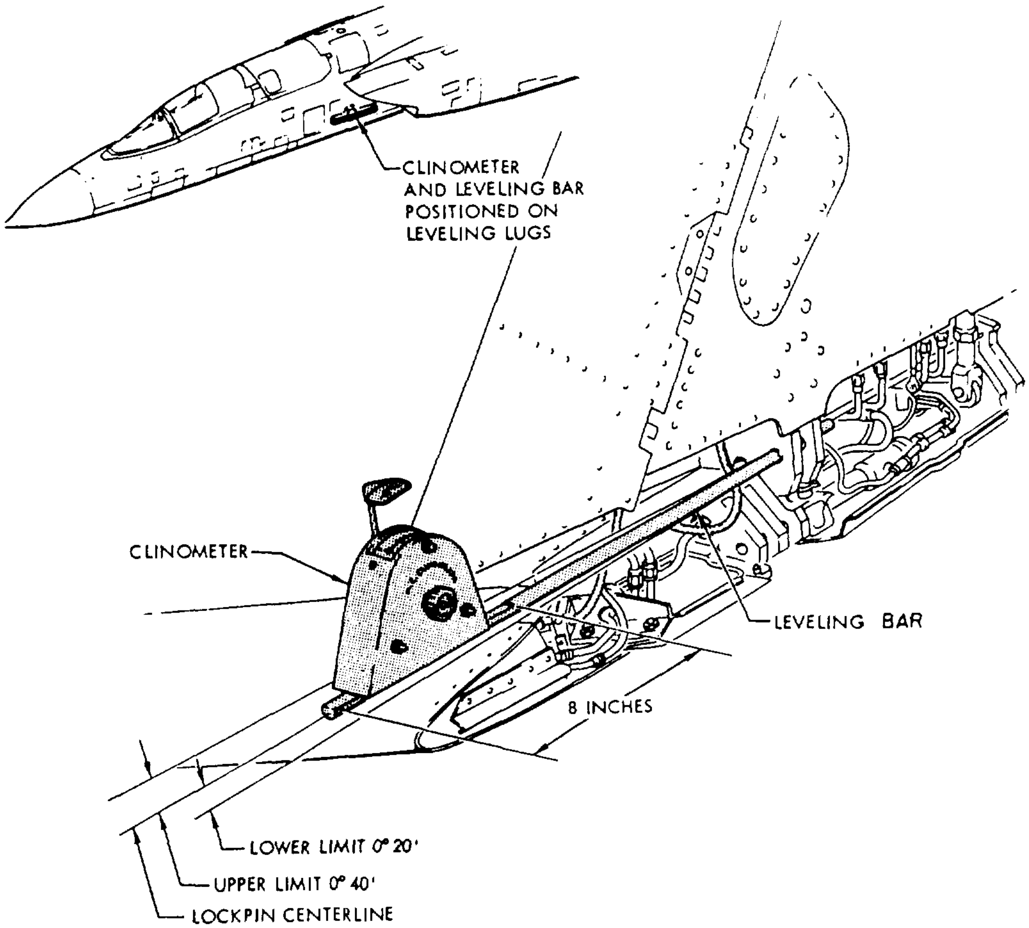


Figure 9-48.—Alignment data—wing twist check.

longitudinal reading, and the upper limit is established by adding 0 degrees, 40 minutes to the longitudinal reading taken in the auxiliary wheel well. For example, if the longitudinal reading was 1 degree, 35 minutes, the lower limit would be 1 degree, 15 minutes, and the upper limit would be 2 degrees, 15 minutes. Figure 9-48 shows a wing twist check on an aircraft. The wing clinometer readings must fall within this range as well as within 0 degree, 12 minutes of each other (right- to left-hand wing readings). This check, together with the steel tape measurements taken when the wings are spread, is a satisfactory check of wing bending and twisting. If the clinometer readings and tape measurements arc not within the tolerances specified, the aircraft must be taken to a depot-level maintenance facility for a complete inspection and final disposition.

RECOMMENDED READING LIST

NOTE: Although the following references were current when this TRAMAN was published, their continued currency cannot be assured. Therefore, you need to be sure that you are studying the latest revision.

General Manual for Structural Repair, NAVAIR 01-1A-1, Commander, Naval Air Systems Command, Washington, D.C., 15 July 1969, Change 11, 15 August 1989, Section II.